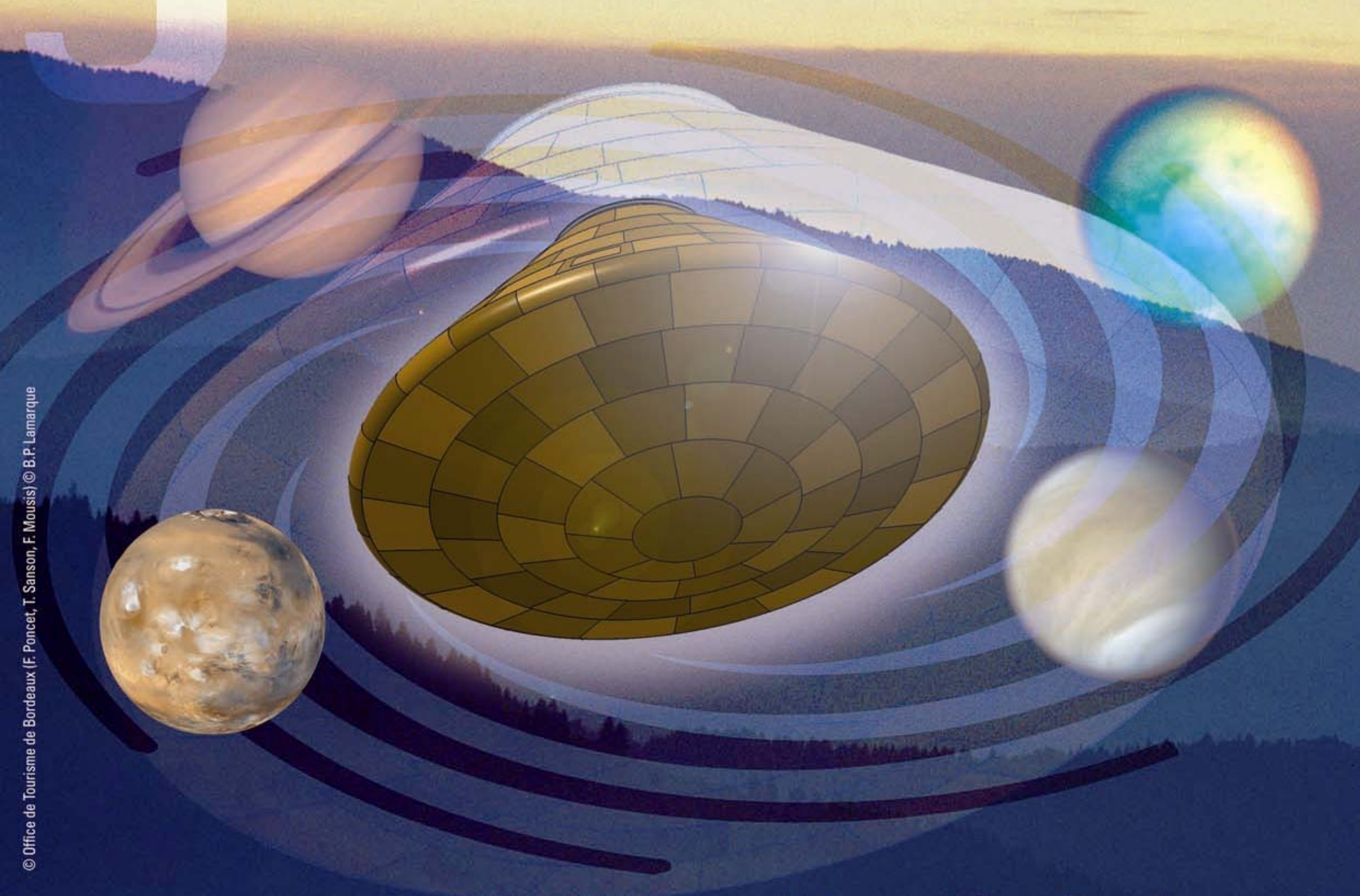
5TH INTERNATIONAL PLANETARY PROBE WORKSHOP

JUNE 25-29, 2007 – BORDEAUX, FRANCE



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BOOK OF ABSTRACTS

Many thanks to IPPW5 Organizers and Sponsors.

PROGRAMME OVERVIEW:

The workshop programme will include invited talks, contributed talks, and posters. The current schedule of sessions with Session Conveners is provided below.

Monday June 25

OPENING, WELCOME

SESSION I: **CURRENT OUTLOOK.** CONVENERS: S. HUBBARD (SETI INSTITUTE) AND J.-P. LEBRETON (ESA/ESTEC)

SESSION II: **MISSION CONCEPT STUDIES AND SCIENCE DRIVERS OF TECHNOLOGY, AND SAMPLE RETURN – VENUS AND MARS.** CONVENERS: K. BAINES (JPL), B. BIENSTOCK (JPL), AND P. PLOTARD (EADS)

Tuesday June 26

SESSION III: **ENTRY, DESCENT AND LANDING CONCEPTS FOR CURRENT AND FUTURE MISSIONS BEYOND EARTH.** CONVENERS: M. WRIGHT (NASA-ARC), A. BALL (OPEN UNIV.), AND W. LEE (JPL)

SESSION IV: **TECHNOLOGY SYSTEMS - ELECTRONICS, INSTRUMENTS & SENSORS, COMMUNICATIONS & BATTERIES.** CONVENERS: P. BEAUCHAMP (JPL) AND TH. BLANCQUAERT (ESA/ESTEC)

BANQUET

Wednesday June 27

SESSION V: **MISSION CONCEPT STUDIES AND SCIENCE DRIVERS OF TECHNOLOGY – GIANT PLANETS AND TITAN.** CONVENERS: A. COUSTENIS (OBSERVATOIRE DE MEUDON) AND T. SPILKER (JPL)

FREE AFTERNOON – SOCIAL ACTIVITIES – EXCURSION TO SAINT-EMILION

Thursday June 28

SESSION VI : **ENTRY, DESCENT, AND LANDING TECHNOLOGIES FOR PLANETARY MISSIONS.** CONVENERS: N. CHEATWOOD (NASA LARC) AND D. LEBLEU (THALES ALENIA SPACE)

SESSION VII: **EMERGING, ENABLING, AND EXTREME ENVIRONMENT TECHNOLOGIES; CROSS-CUTTING TECHNOLOGIES.** CONVENER: L. PELTZ (BOEING)

SESSION VIII: (PART 1) **EARTH ENTRY, DESCENT, AND LANDING FOR SAMPLE RETURN AND CREWED MISSIONS.** CONVENERS: J. ARNOLD (NASA ARC) AND B. FOING (ESA/ESTEC)

Friday June 29

SESSION VIII: (PART 2) **EARTH ENTRY, DESCENT, AND LANDING FOR SAMPLE RETURN AND CREWED MISSIONS.** CONVENERS: J. ARNOLD (NASA ARC) AND B. FOING (ESA/ESTEC)

SESSION IX: **FUTURE OUTLOOK AND CLOSING SESSION.** CONVENERS: J. CUTTS (JPL) AND J.-P. LEBRETON **(ESA/ESTEC)**

POSTERS WILL BE DISPLAYED THROUGHOUT THE WORKSHOP

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IPPW5: Bordeaux, France, 25-29 June 2007

SESSION I: Current outlook
(S. Hubbard, J.-P. Lebreton)
Monday June 25, 8:30-12:00

9:00 - 9:30	**Opening remarks, Welcome, Logistics, etc..**
09:30-10:00	**(Invited) J. Green** *"The NASA's Planetary Science Missions and Plans"*
10:00-10:30	**(Invited) M. Blanc** *"Tentative Title:* ESA's Cosmic Vision*"*
10:30-10:45	**Break**
10:45-11:15	**(Invited) J. Blamont** "*Tentative Title: Ballooning in planetary atmospheres: history and perspectives"*
11:15-11:30	**Q&A and Discussion**
11:30-12:00	**Al Seiff Award Presentation: H. Niemann** *"Planetary Entry Probes and Mass Spectroscopy: Tools and science results from in situ studies of planetary atmospheres and surfaces"*
12:00-13:30	**Lunch**

Monday Lunch offered by: Snecma Propulsion Solide SAFRAN Group EADS astrium

Abstract

Title: NASA's Planetary Science Missions and Plans

Author: James L. Green,
Planetary Science Division
NASA Headquarters

The status and plans of NASA's Planetary Science Division (PSD) program will be reviewed. NASA planetary science missions are either strategic or openly competed through announcements of opportunity and are led by a principal investigator (PI). The competition for the next Discovery and Mars Scout missions are currently underway. Selections for Discovery and Scout will be announced later this year. The New Frontiers program will next be competed in late 2008.

In exploring any particular solar system object, NASA has followed a general paradigm of "flyby, orbit, land, rove, and return." This prescription has been followed most completely for investigations of the Moon and Mars. In contrast, our first flyby or reconnaissance missions include: New Horizons, a mission to Pluto and its large moon Charon (encounter July 2015), Dawn, a mission to Vesta (encounter fall 2011) and then onto Ceres (summer 2015), both in the asteroid belt. Dawn will be launched in the summer of 2007. A complete campaign may not be performed for each interesting object in the solar system, since not all our scientific questions can be studied at all objects, and there are high technological and financial hurdles to overcome for some missions and certain destinations. Moreover, a healthy program of solar system exploration requires a balance between detailed investigations of a particular target and broader reconnaissance of a variety of similar targets. Other new NASA orbiting missions include: Lunar Reconnaissance Orbiter that will orbit the Moon in 2008 in preparations for manned missions, MESSENGER arriving to orbit Mercury in March 2011, and Juno orbiting Jupiter in 2016. The Cassini/Huygens flagship mission has been orbiting Saturn since the summer of 2004 and was recently extended for another 2 years of observations beyond its prime mission that ends in July 2008. The PSD is currently studying Europa, Ganymede, Titan, and Enceladus as potential future flagship missions after Cassini/Huygens.

The PSD has a number of active missions at Mars that include: Mars Reconnaissance Orbiter, Mars Odyssey, and the two mid-sized rovers Spirit and Opportunity. Two new NASA missions to Mars are currently in development: Phoenix and the Mars Science Laboratory (MSL). The Phoenix mission, scheduled for launch in August 2007, is a PI-led mission and the first mission selected through the Mars Scout Program. Phoenix is a fixed Lander designed to measure volatiles (especially water) and complex chemistry (including organic molecules) in the northern polar plains of Mars, where the Mars Odyssey orbiter has discovered evidence of ice-rich soil near the surface. Both Phoenix and MSL missions will be described in more detail.

Last but not least, PSD is proud of its participation on a number of missions from our foreign space agency partners. As NASA develops its strategic plans for future solar system exploration it will continue to embrace international participation and look for opportunities to contribute as appropriate.

ESA Cosmic Vision

by

Michel BLANC

Missing Abstract

Planetary Balloons

by
Jacques Blamont
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Presentation of basic physics of aerostation in planetary atmospheres

- Venus : The Vega balloons. Necessity for altitude excursions a model for a flight from the top to the bottom of the clouds ; a low altitude Lagrangian fleet.
- Mars : heavy payloads (the snake) ; short duration balloons.
- Titan : the RTG montgolfiere.

Planetary Entry Probes and Mass Spectroscopy: Tools and science results from in situ studies of planetary atmospheres and surfaces

Invited Talk for the First Annual Alvin Seiff Award

by

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Probing the atmospheres and surfaces of the planets and their moons with fast moving entry probes has been a very useful and essential technique to obtain in situ or quasi in situ scientific data (ground truth) which could not otherwise be obtained from fly by or orbiter only missions and where balloon, aircraft or lander missions are too complex and costly. Planetary entry probe missions have been conducted successfully on Venus, Mars, Jupiter and Titan after having been first demonstrated in the Earth's atmosphere. Future missions will hopefully also include more entry probe missions back to Venus and to the outer planets.

The success of and science returns from past missions, the need for more and better data, and a continuously advancing technology generate confidence that future missions will be even more successful with respect to science return and technical performance.
The pioneering and tireless work of Al Seiff and his collaborators at the NASA Ames Research Center had provided convincing evidence of the value of entry probe science and how to practically implement flight missions. Even in the most recent missions involving entry probes i.e. Galileo and Cassini/ Huygens Al contributed uniquely to the science results on atmospheric structure, turbulence and temperature on Jupiter and Titan.

SESSION II: Mission concept studies, and science drivers of technology, and sample return – Venus and Mars
(B. Bienstock, K. Baines, P. Plotard)
Monday, June 25, 13:30-17:00

13:30-14:00	**(Invited) J. Cutts, et al.** *"Exploration of Mars, Venus and Titan With Planetary Aerobots: A Pathway to Flight Readiness"*
14:00-14:30	**(Invited) F. Bonnefond** *"Soft-Landing and Hazard Avoidance Aspects for Future Exploration Missions"*
14:30-14:45	**K. Baines, et al.** *"Exploring Venus with Balloons: Science Objectives and Mission Architectures"*
14:45-15:00	**P. Régnier, et al.** *"The MoonTwins Mission Concept: An Affordable and Science Attractive European Mission to Validate MSR Soft Landing and Hazard Avoidance Technologies"*
15:00-15:15	**Break**
15:15-15:35	**(Invited) R. Trautner, et al.** *"The ExoMars Mission - Overview and Expected Results for Planetary Entry Probe Science and Engineering"*
15:35-15:50	**R. Nadalini, et al.** *"HP^3 TEM - A Sensor to Measure the Thermal Characteristics of the Subsurface Temperature of Terrestrial Planets and Asteroids: Results of the Thermal Test Campaign"*
15:50-16:05	**M. Murbach, et al.** *"Atromos - A Mars Companion Mission"*
16:05-16:20	**C. Peterson** *"Mission Architecture Impacts on Venus Surface Probe Lifetime"*
16:20-16:35	**B. Parreira, et al.** *"Hazard Avoidance Techniques for Vision Based Landing: Performance Assessment and Consolidation Analyses"*
16:35-17:15	**Panel Discussion** Topic - TBD
17:15–18:00	**Poster Session II, III, IV, V(authors in attendance)**

EXPLORATION OF MARS, VENUS AND TITAN WITH PLANETARY AEROBOTS: A PATHWAY TO FLIGHT READINESS

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Introduction: With full three-dimensional mobility, aerial robotic vehicles (aerobots) offer many advantages as the scientific exploration platform of choice for the next stage of exploration of Mars, Venus and Titan – the only other worlds in our solar system – that have significant atmospheres as well as solid surfaces. The dense, high-molecular weight atmospheres at Venus and Titan, gentle surface winds, hazardous terrain and scarcity of useable energy make lighter-than-air vehicles the preferred choice for in situ exploration. They enable both prolonged aerial observations as well as in situ surface sampling with little energy usage. They also represent a necessary element of any future surface sample return mission. Lighter than air vehicles have more of a niche role to play at Mars where the thin atmosphere makes aerial or surface deployment challenging and limits the size of payloads at least for first generation systems. The purpose of this paper is to provide an overview of the most current lighter-than-air vehicle concepts and to define the pathway to space flight readiness.

On Titan, surface mobility is compromised by extensive equatorial dune fields, volcanic calderas and a complex of polar hydrocarbon lakes; while aerobots can traverse these areas observing them from close proximity and acquire and analyze samples. Titan's unique atmospheric environment offers two distinct approaches to long-lived aerobots capable of operating for months or even years. First, the low temperature and high atmospheric density enable "hot air balloons" that require only 1% of the heat input of a comparable terrestrial craft. Second, the abundance of methane enables efficient in situ hydrogen replenishment for a light gas balloon or blimp. Titan is in many respects the ideal world for aerial exploration. NASA is currently undertaking a study that considers a Montgolfiere hot air balloon as a key element of a Flagship class Titan mission and will identify key enabling technologies needed to achieve near term flight readiness.

Venus is a more challenging case primarily because of the range of environments. At ~55 km altitude, the atmospheric density and temperature are near earth ambient, but a mist of sulfuric acid is present. The technology is already available to follow up the Soviet VEGA balloon exploration of the early 1980s with extended-duration mission to circumnavigate Venus with superpressure balloons. Both NASA and ESA are considering missions of this type, with further support from JAXA on a mid-cloud balloon. NASA's Venus Exploration Analysis Group (VEXAG) has recommended the study of a Flagship class mission to the surface environment where the temperature approaches 500°C. A lighter-than-air vehicle is a prime candidate.

For Mars, multinational efforts on lighter-than-air vehicles date back almost two decades to a joint Russian French Mars Aerostat mission that was cancelled in 1992. Beginning in the mid-1990s, NASA began an effort to apply the technology of aerial deployment used successfully on the VEGA mission to the tenuous atmosphere of Mars. Stratospheric tests conducted during the last several years and continuing this summer are now bringing the technology for a Mars long duration balloon within reach. One approach is superpressure balloon mission that could fly for weeks in the equatorial regions of Mars; the other is a solar Montgolfiere design that could fly for weeks, but only over the summer polar regions.

Soft-landing and hazard avoidance aspects for future exploration missions

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The solar system exploration by robotic probes has yielded highly valuable science data and greatly improved the knowledge of our immediate space environment. From 1966 successful landings were performed on the moon, Venus and Mars. The surface analysis of planets is a precious complement to in-orbit observation and is essential to prepare for future human exploration of Mars. Two kinds of landing scenarios were used in the past. In the recent Pathfinder and MER missions semi-hard landings were performed where, after a braking phase by parachutes, airbags are inflated around the Lander which allow it to safely bounce onto the surface until it comes to rest. In the soft landing scenario the Lander has propulsive capabilities and achieves a touchdown with quasi zero velocity. In both cases the landing area had to be safe – gentle slopes, small boulders, no cliffs or ridges – in order not to tear the airbags or have the Lander tip over at landing. Areas of scientific interest are however often characterised by erosion or are heavily cratered: rocks, cliffs and rifts are part of the landscape. Landing in such areas therefore excludes the semi-hard landing scenario. Moreover, since the communication delay between the Lander and Earth prohibits real-time monitoring of the descent, autonomous hazard avoidance capability becomes mandatory for future Lander missions. The Lander should be able to reproduce what Apollo astronauts did, i.e. detect hazards on the ground, select a safe landing site that can be different from the nominal one, and perform the necessary trajectory updates to reach the target.

The aim of the present paper is, first, to highlight the evolution of various landing systems with respect to the science needs and to identify the main challenges and drivers for the different solutions. Emphasis is then put on the new generation landing systems with the autonomous hazard avoidance capability in order to explore remote areas. In this case, the Lander maps the terrain during descent and selects a suitable landing site based on the terrain characteristics and a set of predefined criteria. A trajectory planning algorithm then computes a trajectory to reach the selected target. The three hazard avoidance components – hazard mapping, piloting and trajectory planning will be further detailed.

EXPLORING VENUS WITH BALLOONS: SCIENCE OBJECTIVES AND MISSION ARCHITECTURES

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Introduction: Following in the flightpaths of the twin Soviet VEGA Balloons flown in 1985, missions to fly in the high atmosphere of Venus near 55 km altitude have been proposed to both NASA's Discovery Program and ESA's Cosmic Visions. Such missions will answer fundamental science issues promulgated in a variety of high-level NASA–sanctioned science documents in recent years, including the Decadal Study, various NASA roadmaps, and recommendations coming out of the Venus Exploration Analysis Group (VEXAG). Such missions will in particular address fundamental issues of Venus's origin, evolution, and current state, including detailed measurements of Venus's active chemistry and dynamics. As an example of what can be done with a small mission (less than $500M), the Venus Aerostatic-Lift Observatories for in-situ Research (VALOR) Discovery mission will be discussed. This mission intends to fly twin balloon-borne aerostats over temperate and polar latitudes, sampling rare gases, chemicals and dynamics in two distinct latitude regions for several days. A variety of scenarios for the origin, formation, and evolution of Venus will be tested by sampling all the noble gases and their isotopes, especially the heaviest elements never reliably measured previously, xenon and krypton. Riding the gravity and planetary waves of Venus à la the VEGA balloons in 1985, the balloons will sample in particular the chemistry and dynamics of Venus's sulfur-cloud meteorology. Tracked by an array of Earth-based telescopes, zonal, meridional, and vertical winds will be measured with unprecedented precision. Such measurements will help in developing our fundamental understanding of (1) the circulation of Venus, especially its enigmatic super-rotation, (2) the nature of Venus's sulfur cycle, key to Venus's current climate, and (3) how Venus formed and evolved over the aeons.

The MoonTWINS mission concept : an affordable and science attractive European mission to validate MSR soft landing and hazard avoidance technologies

Pascal Régnier[1], Charles Koeck[1] , Philippe Lognonne[2]

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In the frame of its AURORA Exploration programme, ESA has recently initiated several pre-phase A studies aimed at defining MSR precursor mission concepts, whose main objectives are to prepare Europe to take an active role in the future MSR international mission, through the early demonstration of key technologies required to bring back samples of Mars to the Earth in the 2020-2030 time-frame. In addition to the technology demonstration objective, the MSR precursor mission concepts must exhibit a real science interest, and be compliant with an overall cost budget within 400M€. ESA then intends to down-select the best mission concept and submit it for approval at the 2008 Ministerial Council, for a launch as early as 2015, two years after the ExoMars mission. Technologies that should be focused on must complete the ones already endorsed by ExoMars, therefore Planetary Entry, Descent and Soft/Precision Landing, Planetary Ascent, autonomous Rendez-Vous and Docking / Capture are especially targeted. High speed Earth Re-entry was already covered in previous ESA studies. Low Earth orbits, the Moon, Mars, or even large Near Earth Objects or Phobos, are considered by ESA as appropriate mission targets.

In that context, Astrium, with the support of the Institut de Physique du Globe de Paris (IPGP), has conceived an innovative, efficient, affordable and scientifically attractive mission: the MoonTWINS mission. This acronym stands for “Moon Technological Walk-Through and In-situ Network Science”. It consists in launching on Soyuz-Fregat two identical soft landers to the Moon that would first demonstrate autonomous Rendez-Vous in-orbit GNC technologies and operations around the Moon, and then achieve a soft precision landing on the Moon surface with hazard avoidance. This would represent the first opportunity for Europe to validate vision-based and LIDAR technologies that ESA is currently pre-developing through its on-going TRP studies, for preparing future planetary probes missions.

Furthermore the landers carry each a valuable geo-science instruments package, including a high resolution seismometer developed by IPGP, and the targeted landing sites are the Peak of Eternal Light near the Shackleton crater at the South Pole for one lander, and an equivalent site near the North pole. This mission would allow to solve several unanswered questions after the Apollo missions concerning the Moon’s structure and history, and would provide the first insight on the Moon’s deep interior (lower mantel and possible core). It would also represent a major step ahead for Europe in the global Moon exploration program, by potentially achieving the first robotic landing at the site envisioned for the future Manned base. The DLR Institute of Planetary Research is also participating to the definition of the Science objectives and instrumentation for this mission

This paper will first describe the rationale behind the MoonTWINS mission concept; in terms of technology demonstration goals, Moon science and exploration objectives, highlighting its outstanding interest for Europe. Then the mission and system preliminary design is further detailed, including the mission architecture and the spacecraft design. The soft precision landing and hazard avoidance technologies demonstration concepts will be highlighted. A system synthesis conclusion demonstrates the feasibility of the mission at technical and programmatic levels.

THE EXOMARS MISSION - OVERVIEW AND EXPECTED RESULTS FOR PLANETARY ENTRY PROBE SCIENCE AND ENGINEERING

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Introduction: ExoMars is the first Flagship mission in ESA's Aurora Programme [1]. Its aim is to characterise the biological environment on Mars, especially in the shallow subsurface, and to support the preparation for future robotic missions and human exploration. Data from the mission will provide invaluable input for broader studies of exobiology - the search for life on other planets. In the field of planetary entry probe science and engineering, ExoMars will be able to address a number of topics, with measurements performed during the atmospheric entry and descent phase and during the surface mission.

Science and Technology Drivers, Onboard Sensors and Science Payload: The scientific mission and the technology goals of the project are important drivers for the selected Entry, Descent and Landing (EDLS) technology. A set of dedicated sensors will be operated during the atmospheric entry and descent, delivering valuable data on this critical part of the mission. For the landed systems, a payload complement has been selected based on a Payload Confirmation Review process conducted by the European Space Agency. These instruments will perform measurements during the surface phase, and allow a better characterization of the atmospheric conditions on Mars.

A brief overview of the ExoMars project will be presented. The technical and scientific goals and the corresponding drivers for the EDLS phase will be introduced. The available onboard sensors and the selected scientific payload will be presented. The corresponding measurements and the expected results for planetary entry probe science and engineering will be discussed, and the current status of the related project activities will be summarized.

References:

[1] http://www.esa.int/SPECIALS/Aurora/index.html

HP^3 TEM, A SENSOR TO MEASURE THE THERMAL CHARACTERISTICS OF THE SUBSURFACE OF TERRESTRIAL PLANETS AND ASTEROIDS: RESULTS OF THE THERMAL TEST CAMPAIGN.

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The terrestrial planets, many of the major moons and most asteroids in the solar system, are mostly covered by regolith layers, storing in their physical properties a record of the processes having taken place during and after its deposition. The study of these properties below the immediate surface allows a glimpse in the bodies' past for periods of up to billions of years. On Mars, the Viking and MER missions show that already investigations of the first layers of soil are extremely interesting.
Due to the scarcity of missions with landers and the lack on them of instruments able to reach the subsurface, access to depths beyond a few tens of centimetres still remains to be accomplished.
In the last decade, thanks to continuous financial support from ESA and DLR, a new class of lightweight tools has been developed to explore these regions up to depths of a few meters: the Moles (i.e. PLUTO, the sampling Mole on board Beagle 2).
Combining the efficiency of Moles with the most advanced in situ measurement techniques, a fully functional prototype of HP3, the Heat flow and Physical Properties Package has been developed, built and tested for system functionality and performance.
HP3 is an integrated instrument suite, deployed up to 5m in the soil by a dedicated Mole (the "tractor"). While descending, and once in place, HP3 can use all its sensors to measure the regolith's status and properties.
The most important of the three different sensors composing HP3 is TEM, a set of thermometers and heaters used to measures the thermal status and the thermal properties of the soil. A flat Copper-Kapton tether houses the thermal sensors, built with a technology similar to those of conventional foil sensors, and the electrical connection between the payload compartment and the lander element.
Other sensors in the payload compartment help in the navigation of the system and measure the mechanical properties of the soil.
By combining all the data, HP3 can derive the surface heat flow (hence the name). Using the data separately or in some combination, more precise investigations can be done (e.g. the climate of the recent past).
A dedicated test campaign has been conducted in Berlin to investigate the performance of TEM, under simulated space and Martian conditions in one of DLR's thermal-vacuum chambers. To simulate granular regolith, PMMA (Plexiglas) granulate with a mean diameter of 600 µm has been used, for its well known bulk properties and its ease of use.
The test objects, the payload compartment and a representative part of the tether, have been placed in the middle of two cylindrical containers 60 cm long and 20 cm in diameter, equipped with about 50 calibrated temperature sensors and filled with the granulate material.
Two cold plates attached to the ends of the containers were used to control the temperatures inside the sample. Multiple tests have been performed in vacuum, at normal and at Martian atmospheric pressures and at temperatures between ambient and -55°C. The test demonstrated that the TEM prototype functioned flawlessly under all these conditions.
Furthermore, it could be shown that despite of the non-ideal geometry, thermal conductivity measurements with a "Transient Hot Strip"-like arrangement were well reproducible and can be successfully used to determine this parameter with the required accuracy in most tested cases.
The usage of the whole payload compartment as a modified "Transient Hot Wire" to measure the thermal conductivity was also tested with promising results.

ATROMOS – A MARS COMPANION MISSION

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ABSTRACT

The ATROMOS mission proposes to develop two small 8-kg class entry probes on future U.S. and European Mars missions. The effort attempts to solve core issues regarding the development of small piggy-back class missions that have thus far not been successful (e.g., DS-2, Beagle-2) yet could yield very important investigative elements. This is of particular current interest in that most of the planned missions in the next decade are orbiter missions that could be greatly enhanced with the inclusion of a ground element. The initial mission is proposed as a 2 point network that would land one probe on the North Pole and a second one several 100s km south in order to measure key atmospheric gradients. Key to the mission is the development of a new self-stabilizing entry probe (the SCRAMP - Slotted Compression RAMP) that will be flight tested on the SOAREX VI flight experiment in the 4th quarter of 2007. In addition, a simple EDL sequence is proposed requiring a minimum number of pyrotechnic events, making the technology more facile to develop and qualify. Having such an entry system with simple spacecraft interfaces could help 'compartmentalize' riskier technologies/instruments, and greatly enhance the science capability of future orbiter and lander missions. Larger more capable versions of the Atromos mission will be further discussed, with the suggestion that this may lead to future network and other complementary missions.

MISSION ARCHITECTURE IMPACTS ON VENUS SURFACE PROBE LIFETIME

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Introduction: Probes landing on the surface of Venus are subjected to the enormous pressures and temperatures, making survival for any period of time problematic at best. While pressure vessels and thermal control systems are essential to any mission, there are potential architectural options based on taking advantage of available and emerging high temperature components that can extend the lifetime of a Venus lander. These architectures are equally applicable to any system (landed or balloon-based), including sample return, that spends considerable time on or near the surface.

This presentation will cover the relative benefits of different architectures which use primary battery power and passive cooling techniques. The architectures discussed include various combinations of high temperature avionics and telecom components (at both 200 and 500 C along with standard 50 C capable components), advanced thermal protection, high temperature batteries, and beryllium pressure vessel structure. These arhcitectures are evaluated using a simplified operational scenario to determine mission lifetime as a function of total landed mass. While of limited utility in actually predicting the mission lifetime for a particular architecture at a particular mass, the comparison of the results for the different architectures provide a reasonable relative ranking of their effectiveness and can provide guidance on the priority for development of high temperature components.

Hazard Avoidance Techniques For Vision Based Landing.
Performance assessment and consolidation analyses.

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Hazard Avoidance is a key technology for a safe landing of future planetary landing missions. The hazard avoidance function makes use of sensors and computers onboard the lander to detect hazards in the landing zone, autonomously select the most suitable region for landing, and generate the trajectory that retargets the lander to a safer landing site

A HA system is responsible for the detection of any hazards that put in risk the landing mission and path-planning to avoid the identified hazards. Hazard detection implies the lander to be equipped with a proper sensing device. In the frame of this study, an optical sensor, onboard camera, is used to detect hazards (e.g. craters, rocks, boulders, high slopes, etc.) in the landing zone.

In the presented work algorithms are described for vision-based hazard detection, safe site selection and powered landing guidance designed for landing in planets without atmosphere. The algorithms are validated in a realistic simulated scenario representing a landing on Mercury.

The simulation environment includes an image generation module that provides realistic landing scenarios through the generation of an artificial surface representative of the cratered planet. Craters distribution, boulders and terrain roughness can be set to generate different risk levels in the landing area. An image processing function is also implemented being in charge of resolving both: the terrain slope and roughness using “shape from shading” algorithms. These algorithms allow deriving a first mapping of the hazard extracting from the image three basic data set: shadow, texture and slope.

This preliminary risk map is modified through the piloting function that, on real-time, is responsible for evaluating if the current landing site is considered unsafe and, whenever this happens, to provide a new safe LS. Piloting function selects a LS that is, not only ‘safer’ but also reachable. This attainability is addressed considering system requirements and mission constraints such as: available on-board propellant, landing site visibility, propulsion system limitations and guidance algorithms restrictions. Finally a terminal point guidance system, based on the E-Guidance, is also implemented to steer the lander towards the landing site by the piloting module.

Several scenarios have been simulated and the obtained results show a very effective and robust behaviour of the proposed Hazard Avoidance function. The overall retargeting capability, towards a new landing site safer than the nominal one, has been validated with respect to the a-priori known topography.

An exhaustive MonteCarlo campaign has been run assessing the retargeting capability in terms of dispersion and the overall HA robustness with respect to the variation of the most important design parameters. Especially the link between HA performance and onboard navigation has been analysed in detail. Result from this analysis have allowed the improvement of the guidance function within the HA subsystem.

Within this project an important effort has been dedicated towards the real time implementation of the designed and developed algorithms and the optimisation of the most time-consuming nodes within the SW. In this context, the HA SW has been adapted and ported to a LEON computer simulator. Performance analysis of the SW running on the simulator has been preliminary executed. The routines that consumed more resources have been analysed and optimised, and the profiling has been repeated to assess the improvements in performance.

After the optimisation process the computational effort required by the whole Hazard Avoidance function results in 5-6 seconds on a LEON-3 processor (250 Mhz) being the Hazard Mapping (image processing related function) the most demanding process. This performance makes the HA concept suitable for an onboard implementation eventually integrating the most critical process (e.g the Hazard Mapping) on a Field Programmable Gate Array (FPGA).

In a recent consolidation activity, the tailoring of the Hazard Mapping algorithms to the Mars environment has been carried out. Special emphasis has been put on the impact of the sunlight that is scattered through the Mars atmosphere. This makes it to behave as an additional illumination source that has a constant intensity throughout the scenario being observed. Mars atmosphere introduces an additive bias that is introduced as a diffuse illumination (a fraction of the direct illumination, say 1-20%) when rendering the image.

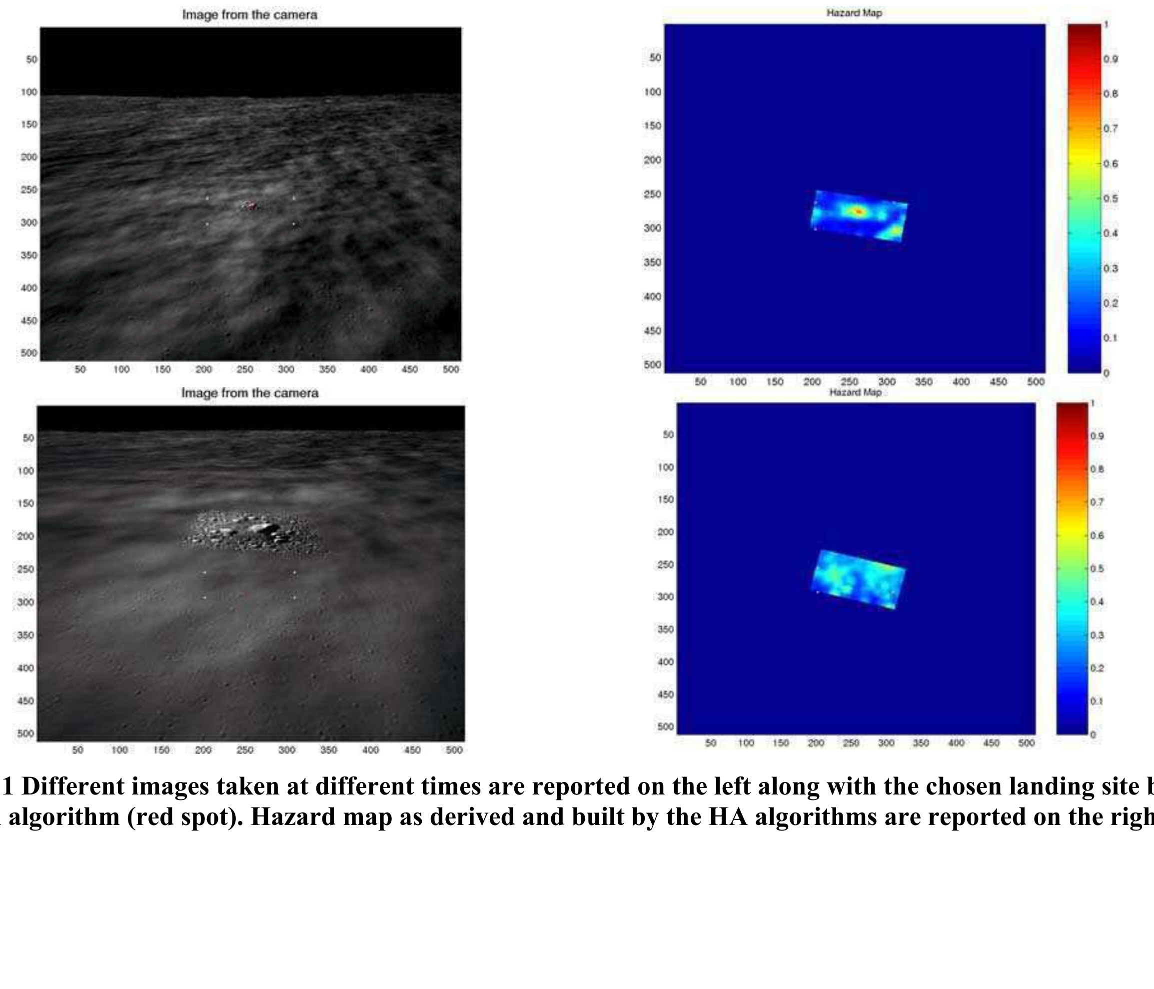

Figure 1 Different images taken at different times are reported on the left along with the chosen landing site by the HA algorithm (red spot). Hazard map as derived and built by the HA algorithms are reported on the right.

The work presented in this paper was done under the ESA/ESTEC contract for the study of Vision Based Relative Navigation Techniques Framework (VBRNAV). (ESA/ESTEC Contract No. 18038/04/NL/JA).

SESSION III: Entry, Descent and Landing Concepts for Current and Future Missions Beyond Earth
(M. Wright, A. Ball, W. Lee)
Tuesday, June 26, 8:30-12:00

8:30-9:00	**(Invited) A.D. Steltzner** *"Mars Science Laboratory Entry, Descent and Landing System"*
9:00-9:30	**(Invited) V. Giorgio, et al.** *"ExoMars Mission and Spacecraft Architecture"*
9:30-9:45	**F. Beziat,** P. Arfi *"A Robust Entry, Descent and Landing System for the ExoMars mission"*
9:45-10:00	**M.J. Gazarik, et al.** *"Overview of the MSL Entry, Descent and Landing Instrumentation (MEDLI) Project"*
10:00-10:15	**Break**
10:15-10:30	**M.R. Grover,** P.N. Desai *"Evolution of the Phoenix EDL System Architecture"*
10:30-10:45	**J.L. Prince, et al.** *"2007 Mars Phoenix Entry, Descent, and Landing Simulation and Modeling Analysis"*
10:45-11:00	**B. Thompson, et al.** *"Design of an Entry System for Cargo Delivery to Mars"*
11:00-11:15	**R. Fisackerly, et al.** *"Entry, Descent and Landing for Mars Sample Return : The European Technology Development and Demonstration Approach"*
11:15-11:30	**C. Wilson,** N.S. Wells *"Balloon-Deployed Probes for Venus"*
11:30-12:00	**Open Mic, Q&A, Panel Discussion, etc**
12:00-13:30	**Lunch**

Mars Science Laboratory Entry, Descent and Landing System

In 2010, the Mars Science Laboratory (MSL) mission will pioneer the next generation of robotic Entry, Descent, and Landing (EDL) systems, by delivering the largest and most capable rover to date to the surface of Mars.   In addition to landing more mass than any other mission to Mars, MSL will also provide scientists with unprecedented access to regions of Mars that have been previously unreachable.   By providing an EDL system capable of landing at altitudes as high as 1 km above the reference areoid, as defined by the Mars Orbiting Laser Altimeter (MOLA) program, MSL will demonstrate sufficient performance to land on a large fraction of the planets surface. By contrast, the highest altitude landing to date on Mars has been the Mars Exploration Rover (MER) MER-B at 1.44 km below the areoid. The coupling of this improved altitude performance with latitude limits as large as 45 degrees off of the equator and a precise delivery to within 10 km of a surface target, will allow the science community to select the MSL landing site from thousands of scientifically interesting possibilities. In meeting these requirements, MSL is pushing the limits of the EDL technologies qualified by the Mars Viking, Mars Pathfinder, and MER missions. This paper discusses the MSL EDL architecture, system and subsystem design and discusses some of the challenges face in delivering such a unprecedented rover payload to the surface of Mars.

EXOMARS Mission and Spacecraft Architecture

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FOREWORD

The Exomars mission is the first **ESA** led robotic mission of the Aurora Programme and combines technology development with investigations of major scientific interest .
The preliminary design of the Mission, Spacecraft and Modules is being performed in the frame of the ongoing phase B1. A System Requirements Review is currently being held between ESA and the Mission Prime Contractor (Alcatel Alenia Space-Italia) to review the top level requirements and the associated preliminary design.
An Implementation Review will then follow to be conducted such that Participating States can decide the final mission configuration, the final Payload Complement and the launch date of the mission. This Irev is presently planned for May 2007.

MISSION DESCRIPTION

Exomars has both technology demonstration and scientific objectives.
The main technology objectives of the mission are the demonstration of the capabilities for safe landing (Entry Descent Landing System) on the Mars surface, mobility (Rover) and access to sub-surface (Drill).
The scientific objectives are:

- to search for signs of past and present life on Mars
- to characterise the water/geochemical environment
- to study the surface environment and identify hazards for the future missions (including human missions)
- to investigate the planet’s deep interior.

The space segment of the mission consists of:

- Carrier Module, which has the function of bringing the Descent Module to Mars through the interplanetary transfer (the Carrier Module is lost after DM separation near Mars)
- Orbiter, accommodating 30-kg, nadir-pointing instrument payload (in the “baseline” scenario this function is performed by a NASA satellite), to operate for a continuous period of at least two Martian years
- Descent Module, including the 20-kg Geophysics/Environment Science Payload (GEP) surface station
- Rover, a semi-autonomous surface mobile platform including the Pasteur scientific payload, to operate for a Nominal Mission of at least 180 sols on the Martian surface. This Nominal Mission shall be completed before the onset of the next global dust storm season..

ExoMars shall be able to land and perform its scientific mission at any longitude, in a latitude band between 15ºS and 45ºN. The mission shall safely deliver the Rover within 25 km (15 km goal) of the target landing site; the Rover, in turn, shall be able to travel at least 25 km on the Martian surface during a six month nominal operations period.
The Rover shall implement a two-way, high-speed communication link with the Orbiter, using every available Orbiter pass to transmit data and receive commands.
An alternative “Direct To Earth (DTE)” two ways link is foreseen for tones transmission and commanding in case of emergency.
ExoMars will also validate the technology for safe Entry, Descent and Landing (EDL) of a large size spacecraft on Mars, the surface mobility and the access to subsurface.

MISSION SCENARIOS

As part of Phase B1, the Prime Contractor was asked to study a number of different scenarios for the ExoMars mission. The basic scenarios are defined as follows.

Baseline scenario
The Baseline scenario features launch in 2013 of a Composite Spacecraft made up of Carrier and Descent Module, using a Soyuz 2-1B launch vehicle. In this scenario the Data Relay Satellite used to support the mission is the NASA Mars Reconnaissance Orbiter (MRO).

Option 1 scenario

The option 1 scenario is the same as the baseline, except for the data relay function, which is performed by a European Mars Telecommunications Orbiter (MTO), launched separately with another Soyuz vehicle. It is under study the possibility to embark a scientific payload as well.

Option 2 scenario

In the option 2 scenario, a composite of Orbiter and Descent Module is launched in 2013 by an Ariane-5 ECA vehicle. The Orbiter performs the carrier functions until delivery of the DM; thereafter it performs the data relay function from Mars orbit. This scenario shall also be available in the option of launch in 2015/2016.

The above basic scenarios allow numerous variants and options as concerns launch window, Earth escape strategy, Mars trajectory, Mars entry and landing, telecom orbit, and so on. This paper outlines the main selection factors. Then, it describes the proposed Soyuz and Ariane 5 mission profiles and their rationale; the relevant spacecraft configurations are as well briefly presented.

A ROBUST ENTRY, DESCENT AND LANDING SYSTEM FOR THE EXOMARS MISSION

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In the frame of the Aurora Exploration Program, ESA initiated industrial studies for the ExoMars mission. Alcatel Alenia Space-France led consortium was awarded the Phase B1 Entry Descent and Landing Systems studies under Alcatel Alenia Space Italy mission prime. The primary objective of Aurora is to create, and then implement, a European long-term plan for the robotic and human exploration of the solar system, with Mars, the Moon and the asteroids as the most likely targets. First mission of the Aurora Exploration Program, ExoMars is scheduled for a 2013 launch. It has the technical objective to demonstrate critical technologies linked to Mars mission, including the critical Entry Descent and Landing phases and the scientific objective at establishing whether life ever existed or is still present on Mars. The EDLS functions are to ensure the Entry, Descent and Landing phases of the Descent Module. This module is housing a Rover carrying an exobiology payload and ensures its safe delivery on the Mars surface.

Based on its comprehensive and world-class successful interplanetary probe experience with Huygens and on its continuous involvement over the past ten years on several projects to Mars, Alcatel Alenia Space-France has designed a robust EDLS concept based on European technologies and presenting high rescaling capability to cope with potential Mission evolutions.

OVERVIEW OF THE MARS SCIENCE LABORATORY (MSL) ENTRY, DESCENT, AND LANDING INSTRUMENTATION (MEDLI) PROJECT

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Abstract

The MEDLI project's main objective is to measure aerothermal environments, sub-surface heat shield material response, vehicle orientation, and atmospheric density for the atmospheric entry through the sensible atmosphere down to heatshield separation of the Mars Science Laboratory (MSL) entry vehicle. The MEDLI instrumentation suite will be installed in the heatshield of the MSL entry vehicle. The acquired data will support future Mars missions by providing measured atmospheric data to validate Mars atmosphere models and clarify the design margins on future Mars missions. MEDLI instrumentation consists of three main subsystems: MEDLI Integrated Sensor Plugs (MISP), Mars Entry Atmospheric Data System (MEADS), and the Sensor Support Electronics (SSE). This paper will provide an overview of the project including the instrumentation design, system architecture, and expected measurement response.

EVOLUTION OF THE PHOENIX EDL SYSTEM ARCHITECTURE

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Introduction: NASA's Phoenix Mars Lander 2007 will begin its journey to Mars from Cape Canveral, Florida in August 2007. But its journey to the launchpad began many years earlier in 1997 as NASA's Mars Surveyor 2001 Lander. In the intervening years, the lander has gone through a series of changes including a cancellation and subsequent rebirth as the Phoenix lander. The systems engineering of the entry, descent and landing (EDL) phase of the Phoenix mission presents a unique challenege of adapting an existing spacecraft system to a new mission scenario, which has resulted in a number of evolutionary EDL arctitectural changes in the course of delivering the Phoenix spacecraft to the launchpad.

EDL Architectural Evolution and Challenges: The EDL architecture of the Phoenix mission has undergone significatant changes in response to changes in arrival conditions, landing site region, as well as through technical insight. The original Mars Surveyor 2001 mission had a more energetic entry and a significantly higher landing site at the equatorial region of Mars. As well, it baselined the use of hypersonic guidance as a demonstration of reduction of landed footprint. Through system risk assessments and complexity trades, Phoenix has been redesigned to perform EDL without hypersonic guidance, and its overall architechure has been reshaped by a lower entry velocity and less stressing landing site elevation at its landing site in the Martian northern latitudes. Technical insights gained during the development and implementation phases of the Phoenix program have resulted in the identification of additional system constraints that have changed the approach to the parachute deployment event, the design of the attitude control system, and the design of powered terminal descent. The reshaping of the Phoenix EDL system design has led to a more robust architecture for the Phoenix EDL event.

2007 MARS PHOENIX ENTRY, DESCENT, AND LANDING SIMULATION AND MODELING ANALYSIS

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Introduction: The 2007 Mars Phoenix lander will launch in August of 2007 before a 10 month cruise to reach the northern plains of Mars in May 2008. Its mission continues NASA's pursuit to find evidence of water on Mars. Phoenix carries upon it a slew of science instruments to study soil and ice samples from the northern region of the planet. In order for these science instruments to be useful, Phoenix must perform a safe entry, descent, and landing (EDL) onto the surface of Mars.

Phoenix EDL Phases: The EDL portion of the Phoenix mission is comprised of many phases. After the 10 month cruise in anticipation for EDL, Phoenix performs a cruise stage separation. Approximately 7 minutes after this separation, Phoenix will enter the atmosphere and begin EDL. From atmospheric interface to parachute deploy Phoenix maintains 3-axis stabilized entry during its hypersonic phase (a phase extending the traditional definition of hypersonic – in this scenario the hypersonic phase ends at parachute deploy at approximately mach 1.5). Fifteen seconds after mortar fire of the parachute, the heatshield is jettisoned. Ten seconds following this event, the lander legs deploy. Reacting to a terminal descent guidance algorithm, the lander separates from the backshell and parachute combination and soon thereafter performes a powered terminal descent to slow the lander enough to land safely on the surface of Mars.

EDL Models: Several Phoenix specific models have been developed to accurately simulate the EDL phase of Phoenix. Models implemented into trajectory simulation during the hypersonic phase include IMU, active hypersonic control system, atmosphere and aerodynamic models. The parachute phase involves a parachute deployment algorithm and inflation model. These parachute models along with a parachute drag model are required to accurately predict the opening loads that the opening of the parachute applies to the parachute as well as to the lander itself. Wind models and atmosphere models significantly impact the effect of the parachute and lander interaction with the Martian environment. In the final phases of EDL, the radar model, terminal descent guidance, propulsive control model are the more important contributions to the success of EDL. In addition, several spacecraft component interaction models are necessary to predict 6dof flight from entry to landing.

Performance Metrics: Phoenix must satisty an array of constraints and derived requirements in order to successfully perform EDL. Trajectories are simulated and submitted through a probabilistic Monte Carlo analysis to determine the risks associated with the Phoenix descent. Analysis of these performance metrics will be described in detail in this paper. Some examples of performance metrics are, but are not limited to, the following observations. Phoenix must maintain within required attitude and attitude deadbands for proper functionality of spacecraft systems. Parachute opening loads must be within set requirement for the lander and the parachute to sustain. For all components of EDL to perform correctly, a sufficient timeline between parachute deploy and lander separation must be achieved. Requirements on parameters at landing limit the amount of vertical and horizontal velocity that the spacecraft may endure upon landing on the surface. Of upmost importance to landing site selection is the footprint, or landing ellipse, of probabilistic predicted landing latitudes and longitudes of the spacecraft. The size of the landing ellipse determines regions of scientific interest that may or may not be achievable. Once all engineering and scientific requirements are met pre-flight, Phoenix can successfully demonstrate another entry, descent, and landing on the surface of Mars.

DESIGN OF AN ENTRY SYSTEM FOR CARGO DELIVERY TO MARS

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Abstract: Human missions to Mars will require the supply of consumables such as food, water, and oxygen, in addition to other equipment and resources that the astronauts may not necessarily be able to bring with them. A sustained campaign of Mars exploration, in which astronauts are on the surface for months to years at a time, may require regular supply missions, in a similar manner as the International Space Station requires regular Progress Module or Space Shuttle resupply missions.

This paper outlines a systems study for an entry vehicle for human resupply cargo delivery to Mars. The top-level requirements for such a mission are to deliver 20 metric tons of human resupply cargo to the surface of Mars at 0 km altitude (MOLA reference) with a landed accuracy of less than 1 km. The system level trade studies and configurations considered are discussed and a baseline configuration that satisfies the top-level requirements is presented. The nominal concept of operations includes aerocapture with an inflatable aerodynamic decelerator, hypersonic entry with a sphere-cone aeroshell, supersonic deceleration with another inflatable aerodynamic decelerator, and terminal descent using chemical propulsion. Vehicle analysis includes subsystem mass estimation, propulsion sizing, trajectory simulation, aerothermal analysis, thermal protection system sizing, and cost estimation. Uncertainty analysis is performed through Monte Carlo simulation, and the vehicle is sized to achieve the mission requirements to at least a 99% confidence. Uncertainty in atmospheric density, vehicle mass, aerodynamics, and entry state are analyzed.

The technologies necessary to meet the top-level mission requirements are outlined and their current state of development is discussed. Systems that will require technical development in order to accomplish the mission objectives include inflatable aerodynamic decelerators, supersonic propulsion, and in-space assembly of a thermal protection system. Furthermore, facilities and procedures to test and qualify these systems will need to be developed to enable such a mission. An estimate of development and production cost is performed.

ENTRY, DESCENT AND LANDING FOR MARS SAMPLE RETURN: THE EUROPEAN TECHNOLOGY DEVELOPMENT AND DEMONSTRATION APPROACH

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The Mars Sample Return (MSR) mission represents one of the major milestones in the future exploration of Mars and of the solar system in general. Providing unparalleled opportunities for scientific investigation, in particular toward the search for past or present life, the MSR mission also involves many key enabling capabilities for future exploration missions including future human missions to the red planet. Europe has identified the MSR mission as one of its major space exploration priorities and the European Space Agency (ESA) is pursuing the development of key technologies for MSR through the European Space Exploration Programme - Aurora. This paper presents the ESA approach to the development of one of the key enabling capablities for MSR: Enty, Descent and Soft-Precision landing. This approach is built on several major elements including: past and ongoing MSR mission system studies; progressive and focused technology development activities; and the pursuit of intermediate technology demonstration mission concepts.

Demonstration mission concepts include, for example, a soft-precision lander on the Moon which would employ an autonomous guidance, navigation and control (GNC) system to ensure a safe and accurate landing at a topographically challenging location on the lunar surface. The technologies needed to enable the hazard avoidance required include camera-based visual sensors, laser-based LIDAR sensors, as well as the required avionics and algorithms. The development and testing approaches to each of these technologies, as well as their integration with the overall system development, will be articulated here.

Balloon-deployed probes for Venus

Colin Wilson (Oxford University) & Nigel Wells (Qinetiq Ltd)

The vertical profiles obtained from dropping probes are always of great interest for the study of planetary atmospheres. This is especially true for Venus, where the opacity of the atmosphere renders remote sensing of the deep atmosphere impossible at all but a few wavelengths. In the context of the European Venus Explorer (EVE) mission currently being proposed to ESA, we have considered two versions of balloon-deployed probes: (1) small, 100g ‘micro-probes’, and larger 1kg ‘imaging probes’. In both cases, we have assumed that the probes communicate to the balloon, rather than directly with an orbiter; and that this communications link is also used to determine the location of the probes.

The first of these options was studied by Qinetiq under contract to ESA for its ‘Venus Entry Probe’ Technology Reference Study. These 100g microprobes would have insufficient thermal mass to keep the central electronics below 125 ºC, the maximum temperature for most commercial electronics, below altitudes of ~10 km. These probes would thus focus on atmospheric dynamics and radiative balance, with a 10g science payload of pressure, temperature, and light flux sensors.

In a second study with Qinetiq, we have studied the feasibility of larger, 1 kg balloon-deployed probes. The main driver behind these probes was to give the probes sufficient thermal mass to reach the surface, permitting some surface science goals to be addressed. With the larger mass, the payload can include a camera for imaging the surface, as well as a simple chemical payload to enable chemical profiling right down to the surface. The surface science goals of these probes would be (1) to identify surface mineralogy at the landing site by mapping surface reflectivity data at a few wavelengths in the 0.7 – 1.0 µm range, and (2) to obtain images of surface morphology at high spatial resolution.

The trade-offs identified in the studies will be discussed.

SESSION IV: Technology Systems, Electronics, Instruments and Sensors, Communications and Batteries
(P. Beauchamp, Th. Blancquaert)
Tuesday, June 26, 13:30-17:00

13:30-14:00	**(Invited) T.S. Balint, et al** *"Extreme environments Technologies for Probes to Venus and Jupiter"*
14:00-14:15	**K. Andrews,** D. Divsalar, S. Dolinar, F. Pollara *"Radiation Tolerance and Information Theory"*
14:15–14:30	**J.-J. Berthelier,** M. Godefroy, M. Hamelin, E. Seran, F. Simões, S. Yahi, CETP/IPSL, and the ARES team, *"Balloon Flight Validation of ARES, an Atmospheric Electricity Instrument proposed for the ExoMars-GEP"*
14:30–14:45	**R.J. Brill**, T.N. Le, P. Papadopoulos, *"Development and Characterization of a compact Sun Photometer for Planetary Applications"*
14:45–15:00	**S. Sheridan**, S.J.Barber, G.H.Morgan, D.Morse, I.P.Wright, *"Miniaturized mass spectrometry for future space flight applications"*
15:00–15:15	**D. T. Young** & J. Hunter Waite, *"Ultra-high resolution Mass Spectrometry for Planetary Probes"*
15:15–15:30	**Break**
15:30–16:00	**(Invited) Brian J. Drouin**, John C. Pearson, Frank W. Maiwald, *"Rotational Spectroscopy on Planetary Probes"*
16:00–16:15	**V. Lazic** (ENEA), et al. *"Laser induced Breakdown Spectroscopy of soils, rocks and ice at subzero temperatures in simulated Martian conditions"*
16:15–16:30	**J. Apa**, J.M. Lafleur, and J.L. Sharma, "*Modification of Deep Space 2 Mars Microprobes for Earth Asteroid Investigation"*
16:30–16:45	**B. White**, et al. *"SOAREX-VI Re-entry Flight Test Experiment – Electronic Systems of the Slotted Compression Ramp (SCRAMP) Probe"*
16:45–17:00	**K.-M. Cheung,** *"Probabilistic Instrument Data Generation Analysis Using a Variant of Saddle-Point Approximation"*
17:00–18:00	**Poster Session VI, VII, VIII (authors in attendance)**

19:30-23:00	**Workshop Banquet** *(precise timing to be confirmed)*

Banquet Dinner offered by : ThalesAlenia Space — A Thales / Finmeccanica Company

Oenological Animation offered by : cea

EXTREME ENVIRONMENTS TECHNOLOGIES FOR PROBES TO VENUS AND JUPITER

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Introduction: NASA's 2006 Solar System Exploration (SSE) Roadmap, in line with National Research Council (NRC) recommendations, identified the in-situ exploration of Venus and the Giant Planets as high priority science objectives. Among the recommended missions, the Flagship class Venus Mobile Explorer (VME) mission would encounter temperature and pressure conditions as high as ~460°C and ~90 bar near or at the surface of Venus. At Jupiter, a New Frontiers class atmospheric deep entry probes mission is expected to descend to ~100 bar – which is ~250 km from the 1 bar pressure elevation – where the temperature reaches ~500°C. Operating times for these proposed missions are mission architecture dependent, and can vary from about an hour for Jupiter probes to over 90 days for VME. In this paper we address technologies for extreme environments at these targets, with a focus on pressure and temperature mitigation over planned mission lifetimes. Pressure vessel designs are outlined from the current State of Practice (SoP) to advanced concepts for proposed missions over the next decades. Thermal management and design – both active and passive – are also addressed. Furthermore, we will discuss other enabling technologies, including high temperature electronics, power storage and power generation; and system architecture options, such as hybrid systems. It is expected that the findings from this assessment – documented in detail in NASA's Extreme Environments Technologies report – would help NASA with identifying future technology investment areas, and in turn enable or enhance planned SSE missions, while reducing mission cost and risk.

Radiation Tolerance and Information Theory

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Several future missions will operate in medium to high radiation environments, requiring technologies or strategies that could extend their capabilities for reliable operation in these extreme environments.

The current situation of radiation hardening technology is reminiscent of the state of communication technology before Claude Shannon illustrated the principles of information theory and coding for communication. It was then believed that the only way to increase communication range, rate, or reliability was to increase the transmitter power or antenna gain. Rudimentary repetition codes had been suggested but they didn't provide an effective overall improvement.

Shannon's idea was instead to operate the system in a regime where it makes many errors and eliminate them through the efficient introduction of controlled redundancy. The existence and practical implementation of codes that accomplish this has made an enormous impact on communication systems. Our approach to radiation hardening is similar: to devise efficient introduction of redundancy to protect storage and computation devices.

Coding theory provides powerful methods to make storage devices more reliable to both transient errors, caused by single event upsets, and to permanent damage due to massive radiation effects. Error correcting/detecting codes for memories have been in use for many years and simple redundancy schemes for entire storage device failure are also well known. Our methods extend the current state-of-the-art in several directions: error correcting codes are not used just during normal read and write operations, but they are repeatedly used to “scrub” the memory from any errors present; permanently compromised memory cells are treated as “erasures” or kept in a suitably compressed list of “bad” cells not to be used again; entire memory block failures are dealt with improved RAID-like systems based on a redundancy scheme provided by powerful error correcting codes. The overall result is vastly improved reliability at the expense of a very modest increase in device size and processing power, without having to resort to conventional, expensive radiation hardening techniques.

Such concepts have the potential to revolutionize instrument capabilities in harsh environments, by increasing the size and reliability of affordable storage and allowing much simpler and lower cost methods for transmission of the scientific data.

BALLOON FLIGHT VALIDATION OF ARES, AN ATMOSPHERIC ELECTRICITY INSTRUMENT PROPOSED FOR THE EXOMARS-GEP.

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and the ARES team

Introduction: The Atmospheric Relaxation and Electric field Sensor (ARES) experiment on the Geophysical and Environmental Package (GEP) of the EXOMARS project is devoted to the investigation of atmospheric electricity on the surface of Mars. The instrument will measure the local conductivity of the atmosphere and detect electromagnetic waves that reveal planetary scale characteristics. On Earth, the stratosphere offers in some aspects similar conditions than on the surface of Mars: low pressure, ionization processes, natural waves and DC electric fields in the ionospheric cavity. Then balloon borne experiments are the best mean to test the prototypes and validate the instrumental design. Balloon flights also provide the harsh environment – temperature variations, electromagnetic interferences – to test the instrument in operational conditions. During the AMMA (Africa Monsoon Multidisciplinary Analysis) campaign the AIRS prototype of ARES flew onboard a stratospheric balloon launched from Niamey (Niger), August 7, 2006 up to a ceiling altitude of 23 km. After a short presentation of ARES and its prototype AIRS we discuss the main scientific and technical results of the experiment.

The ARES and AIRS instruments: ARES is a double probe electric field instrument with two cylindrical sensors installed on the meteorological mast of the GEP. AIRS is similar but with smaller separation between electrodes and the electronics is adapted to balloon flight requirements. The instrument measures the magnitude of the vertical component of the electric field and the potential of GEP (or structure for AIRS) with respect to background from DC to 2 kHz and up to ~ 250 V/m. In a dedicated mode, the detection range can be increased to ~20 kV/m. The vertical electric component of electromagnetic waves and the AC fluctuations of the potential of the reference body are measured in the frequency range from 8 Hz to 4 kHz. Operated in the relaxation probe mode, the instrument provides a measurement of the atmospheric conductivity separately for positive and negative ions. With DC field and conductivity measurements, ARES/AIRS allows to study quiet areas of the global circuit as well as local strong fields generated by charged clouds and thunderstorms, or dust storms especially in the case of Mars. One of the challenges of ARES is the ability to measure intense electric fields using floating reference amplifiers. The wave measurements provide information about large scale characteristics of the ionospheric cavity.

The AIRS results after the Niamey 2006 balloon flight: During the ascent the balloon was far from any mesoscale convective system but the DC atmospheric electric field and conductivity profiles were slightly disturbed. Small scale electric field variations and turbulence phenomena were observed from ~ 11 km up to the tropopause at 16.5 km, which can be interpreted as resulting from the crossing of charged cirrus clouds. The same kind of structures was also unexpectedly observed at higher altitudes, which deserves data correlation with other instrument. Numerous lightning events were detected by the optical sensors in correlation with AC bursts.
The relaxation probe worked nominally, the relaxation curves presented the expected shape, and the derived conductivity is ~10% larger for negative ions, as expected. However, conductivity magnitude is lower than the reference values usually found in this altitude and latitude ranges. The reason for this discrepancy is under investigation.
Although the electromagnetic noise was significant, mainly during the first stage of the flight, the instrument was able to detect, at least, the three lowest Schumann resonances.

Technical results: The instrument measured DC and AC electric fields of several tens of V/m, validating the floating ampllifiers concept. The very high voltage mode has not been tested only by lack of a thunderstorm on the path.

DEVELOPMENT AND CHARACTERIZATION OF A COMPACT SUN PHOTOMETER FOR PLANETARY APPLICATIONS.

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Introduction: Understanding the atmospheric physics and chemistry of a planetary atmosphere, as well as modeling its climatology, requires a detailed knowledge of the composition and phase of atmospheric gasses, size and distribution aerosols, and the upwelling and down welling energy flux. Long-term measurements of planetary atmospheres are necessary since the atmosphere is continually changing, varying both, spatially and temporally.

Planetary orbiters are most often used to obtain these measurements globally from outside the atmosphere. However the revisit times/rates of the orbiters are constrained by their orbital parameters. Ground based instruments, mounted on landers and rovers, can provide complimentary data to orbital measurements. Ground measurements can provide validation during orbiter over flight, while also providing continuous measurements throughout the diurnal and seasonal planetary cycles. If these instruments are mounted on atmospheric vehicles, such as airplanes or balloons, they can provide as detailed atmospheric profile.

Concept: The concept is to develop a lightweight, compact, highly reliable sun photometer suitable for use in planetary environments. Most sun photometers incorporate a rigid tracking mechanism to follow the sun. The tracking mechanism may provide a point of failure during the entry, descent, and landing phase or later during the operational phase due to the harsh entry/planetary environment. Making these systems more robust usually requires increasing the instruments size, weight, and/or power requirements.

The instrument described here will use an optical system to provide a hemispherical field of view removing the need for a sun tracking mechanism. A CCD array is placed at the base of the optical system captures and records the light. Measurements at specific wavelengths are achieved by interposing various interference filters into the light beam between the optics and the CCD array by means of a filter wheel.

Current work: Although the CCD array records both the diffuse light entering the cone as well as the direct sun beam, current efforts are solely focused on validating the direct beam measurements. Efforts have been primarily focused on characterizing the instrument angular response, as well as validating measurements of aerosol optical depth, and water-vapor columnar abundance. Measurements made in the field are compared with a Microtops II handheld sun photometer. The spectral response of optical system, CCD array and filters have been characterized in the laboratory using an Ocean Optics USB4000 spectrometer and either an Ocean Optics LS-1 or a 1,000 W FEL lamp light source.

MINIATURISED MASS SPECTROMETRY FOR FUTURE SPACE FLIGHT APPLICATIONS

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Mass spectrometry is one of the most powerful and widely applicable analytical techniques available to planetary scientists, and mass spectrometers of various types have traditionally formed the heart of many spacecraft orbiter, probe and lander payloads. However, they have also been relatively resource-intensive, placing substantial mass and power constraints on mission architectures, putting mass spectrometry beyond the reach of some smaller mission payloads. Recent advances in ionisation, analyser and detection components allow smaller instruments to be built bringing mass spectrometry into the reach of smaller spacecraft platforms. Taking two recent examples, the 4.5 kg GC/MS Ptolemy on the Rosetta lander Philae[1], and the 6.5 kg Gas Analysis Package on Beagle2[2] as a starting point we will outline the status of a new generation of future instrumentation, including a 2 kg dust composition detector for a Europa orbiter mission proposal, a 500 g device to analyse volatiles in permanently shaded Lunar regions, a 150 g instrument designed to be carried on a mole or melting probe[3], and a sub-100 gram mass spectrometer for use in planetary microprobes.

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ULTRA-HIGH RESOLUTION MASS SPECTROMETRY FOR PLANETARY PROBES

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Mass spectrometry is probably the most general and precise analytical technique that can be brought to bear on the detection and analysis of organic and inorganic substances. Mass spectrometers can also be combined with chemical separation techniques such as gas chromatography to further enhance selectivity and sensitivity for compounds such as biomarkers, which are of crucial importance to exploration of the planets and the search for past or extant life. Another advantage of mass spectrometry is that samples can be acquired directly from planetary atmospheres or through any number of sample preparation and extraction techniques such as pyrolysis or laser ablation. Despite these advantages, and for all their power as analytical tools, chromatographs and mass spectrometers would be irrelevant for in situ planetary research unless they could be made rugged, small, lightweight and relatively low power. In this paper we briefly review the history of mass spectrometry applications in space flight and then describe development of a small (~40 cm) time-of-flight mass spectrometer with demonstrated mass resolution well above 10,000. We will also discuss the use of 2-dimensional gas chromatography as well as techniques for adapting both instruments to planetary probes.

ROTATIONAL SPECTROSCOPY ON PLANETARY PROBES

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High resolution spectroscopy at long wavelengths has been widely used for characterization of gas-phase species. It is particularly powerful because of the ability to absolutely specify the carrier of the (usually rotational) spectrum, including the isomer, conformation and isotopic content. The technique has not enjoyed mainstream analytical use due to technological constraints on the source and detector systems and therefore it has been relegated primarily to use in research laboratories. At JPL we have incorporated new source technology developed for Herschel and commercially available room temperature detectors into the JPL millimeter and submillimeter spectrometer and demonstrated its use for wideband detection of trace gas-phase species. This new system can be configured in a compact, portable design for in-situ measurements. The spectrometer has been tested with a laboratory simulation of the activated Titan atmosphere and shown to be sensitive to trace species such as methyl cyanide, hydrogen isocyanide, ammonia and the cyanide radical. The list of detectable species is essentially all gas-phase molecules with an electric or magnetic dipole moment. Current detection efficiencies require an absorption pathlength of approximately five meters for a suitable cadre of interesting species to be rapidly detected, i.e. millisecond integration times. The demonstrated room-temperature technique is currently five orders of magnitude above the theoretical detection limit.

LASER INDUCED BREAKDOWN SPECTROSCOPY OF SOILS, ROCKS AND ICE AT SUBZERO TEMPERATURES IN SIMULATED MARTIAN CONDITIONS

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The future missions to Mars of NASA (Mars Science Laboratory) and ESA (Exomars) plan to include a LIBS instruments to analyzes the elemental composition of the surface materials. A presence of a moisture in the martian soils has been already detected, as well as surface and subsurfacewater-ice. Both the water and ice can affect the instrument analytical capabilities.

We applied LIBS technique on the slightly moisted soil/rock samples in simulated Martian conditions. The signal behavior as a function of the surface temperature in the range from +30°C to -60 °C was studied for the first time. We observed the strong signal oscillations below 0°C with different negative peaks, whose position, width and magnitude depend on the surface granulometry and roughness. In some cases, the signal was reduced for one order of magnitude with consequences for the LIBS analytical capability. Such a signal behavior is attributed to the presence of supercooled water inside the surface pores [1] or on the interfaces [2], which freezing point depends on the pore size [2-3]. On a same rock sample with different grades of the surface polishing, the signal shows a different temperature dependence. Its decrease was always registered close to 0°C, corresponding to the freezing/melting of normal disordered ice. The latter can be present inside larger pores and scratching. An amount of the signal reduction at the phase transition temperatures does not seem to change with the laser energy density in the examined range. Comparative measurements were performed on a slowly cooled water solution (i.e. in a presence of supercooled water between the ice crystallites). The large LIBS signal depressions were observed close to the temperatures of the known supercooled water transitions. The same negative peaks, but with the smaller magnitudes, were also registered on some rock/soil samples containing the moisture.

Although here described measurements were performed in CO2 environment at pressure of 7 mbar, similar effects can be expected in other environments above the triple point, whenever the moisture is present in the sample. From these results, it can be suggested that LIBS analysis also in Martian conditions, should exclude the operation close to the points of the known water/ice transition, i.e. around 0°C, - 40°C and -50°C, where the ablation rate and the signal are strongly reduced. Particularly, the phase transition close to -50 °C produced a severe signal reduction, for 80% on one rock sample and even for two orders of magnitudes on the ice. Very small ablation rate of the ice at this temperature can compromise also a possibility to remove it from the surface before analysis of underlying soil/rock surface. The larger focal spot size would allow for interaction with a wider range of pore dimensions present on the natural samples, thus to reduce the signal changes with the temperature due to water/ice transitions dependent on the pore size.

This work point out a necessity to examine further LIBS signal behavior on moist samples at subzero temperatures, including the influence of pore size distribution, surface roughness, degree of moisture and pore filling and their consequences on the calibration for quantitative sample analyses. In addition, it indicates that the calibration procedure should take into account the plasma parameters, as the measured electron density results dependent on the surface temperature.

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MODIFICATION OF DEEP SPACE 2 MARS MICROPROBES FOR NEAR-EARTH ASTEROID INVESTIGATION

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One emerging goal for planetary exploration, especially in the U.S. Mars robotic program, is the extensibility of technologies to broader space applications. For instance, future robotic Mars missions will likely serve as testbeds for technologies needed for human exploration. In addition, the Mars Technology Program at NASA JPL includes several base technology focus areas that are applicable not only to Mars missions, but planetary exploration in general. In line with this emphasis, this paper presents a modification of the 1999 Mars Microprobes that allows substantial scientific return for the case of a proposed mission to the near-Earth asteroid Apophis.

Goals of the baseline Discovery-class mission of which the probes are a part address NASA Solar System Exploration Roadmap objectives on solar system origins and hazard characterization. One important advantage to the use of impactor probes on Apophis is the large amount of surface and subsurface compositional knowledge that can be derived from minimal instrumentation and mass. To adapt the Mars Microprobes to the environment of Apophis, several modifications are proposed. The basic forebody-aftbody configuration is preserved. The original soil water experiment is removed and a two-axis, high-sensitivity accelerometer to measure seismic effects of subsequent probe impacts is added. The final instrumentation package also includes a descent accelerometer, impact accelerometer, and thermal conductivity experiment, which existed in the original microprobe design. Accelerometers provide impact deceleration profiles, which can be used to derive surface composition and porosity, and temperature sensors track the temperature gradient in order to determine thermal properties of the inner structure.

The baseline Apophis mission involves the sequential launch of four probes over a period of two weeks from an altitude of approximately 400 m. Launch is executed via an added propulsive module consisting of a cold-gas nitrogen thruster. Impact occurs at 100 m/s after initial pointing and spin-up while housed in the main spacecraft bus. Upon impact, ejecta and crater formation are monitored by the main spacecraft to allow further estimates of composition. The design of this architecture allows for significant flexibility, which is useful in the highly uncertain environments often encountered in planetary exploration. For instance, if the first impactor fails, the aforementioned 100 m/s impact velocity may be reduced for subsequent launches to increase probability of success.

Specific engineering design issues addressed in this paper include thermal considerations and impact effects (e.g. expected penetration depth, deceleration, crater size, and ejecta properties). Also included in this paper is a brief background of similar previous design proposals. Finally, it is argued that this modified design is applicable to investigation of moons and minor bodies other than Apophis.

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SOAREX-VI RE-ENTRY FLIGHT TEST EXPERIMENT - ELECTRONIC SYSTEMS OF THE SLOTTED COMPRESSION RAMP (SCRAMP) PROBE

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ABSTRACT

A discussion of the electronic systems used in the Slotted Compression Ramp (SCRAMP) probe, a part of the SOAREX-VI (Sub-Orbital Aerodynamic Re-entry EXperiment) flight test scheduled for launch on the ATK ALV-X1 sub-orbital flight in late 2007. The SCRAMP is the principle entry body of the SOAREX-VI experiment; a uniquely self-stabilizing re-entry probe with a cylindrical forebody, hemispherical nosecap and aft flare. It is powered on via a separation loop cut by a pyrotechnic cable cutter, initiating RF transmission of sensor data. Sensors include pressure transducers, angular rate sensors, thermocouples, accelerometers, color video, and advanced TPS recession sensors. All analog data are converted to digital by a PCM encoder. As no post-mission recovery is planned, A C-band radar transponder is used to transmit probe velocity data, with all other data transmitted to various ground assets and/or the TDRSS satellite network via dual S-band transmitters with custom antennas. Data will be used to characterize pressures at the probe nose as well as the shock-shock interaction at the probe flare, external probe temperature and internal sensor calibration temperatures, acceleration in three axes, rotation and probe stability, as well as thermal protection system recession rate and amount. The data gathered will help validate the design of this class of highly stable probe for use in future planetary missions.

Probabilistic Instrument Data Generation Analysis Using a Variant of Saddle-Point Approximation[1]

(Approximating the Tail Probability of Sum of Independent Random Variables)

Kar-Ming Cheung & TBD

April 19, 2007

Abstract

This paper describes a probabilistic instrument data generation analysis method using a variant of Saddle-Point approximation technique, which is useful for spacecraft or probes equipped with multiple sensor instruments that generate data in a non-deterministic way. This is particularly true for onboard instruments that employ data compression to reduce transmission bandwidth. As data compression, whether lossless or lossy, is a fixed-to-variable data conversion process depending on the data content and the compression algorithm used. Current analysis approaches either involve extensive simulations which are time-consuming, or assume that the statistics of various data sources are "reasonably similar to Gaussian", and use Gaussian approximation to simplify the analysis. However experience from prior missions indicates that this is not quite true, especially for spacecraft or probe with a small number of instruments, and/or when the statistics of individual instrument data sources are far from Gaussian. The Saddle-Point approximation technique does not resort to the Gaussian assumption, and greatly improve the accuracy of estimating instrument data generation for a mission. This in turn provides accurate estimates to optimize spacecraft design in the areas of onboard storage, bus bandwidth, downlink capabilities, etc. This technique can also be applied to a wide range of engineering problems including link analysis, queue overflow probabilities for different traffic models, and mass, power, and cost estimations for spacecraft design.

[1] This research was funded by the Architecture Modeling and Simulation Effort of the SCaN Constellation Integration Program. In this paper we apply the technique to the problem of instrument data generation of spacecraft.

SESSION V: Mission Concept Studies and Science Drivers of Technology- Giant Planets and Titan
(A. Coustenis, T. Spilker)
Wednesday, June 27, 2007, 8:30-12:00

8:30-8:50	(**Invited**) **S. Atreya, S. Bolton, A. Coustenis, D. Gautier, T. Guillot, P. Mahaffy, B. Marty, H. Niemann, T. Owen** *"Formation of giant planets and their atmospheres: entry probes into Saturn and Beyond"*
8:50-9:05	**B. Marty, T. Guillot, A. Coustenis, and the KRONOS Team** *"Kronos - Saturn exploration with probes"*
9:05-9:20	**M. Ayre, P. Regnier, J.-M. Bouilly, S. Kemble** *"Mission Design Consequences of a Jovian Atmospheric Probe"*
9:20-9:40	**T. Balint, et al.** *"JPL's studies of Mission to Saturn with Probes"*
9:40-9:55	**P. Subrahmanyam and P. Papadopoulos** *"Aerothermal Analysis for Planetary Entry Probes Using DOTNET Framework and OLAP Cubes database"*
9:55-10:15	**Break**
10:15-10:30	**H. Niemann** *"The Huygens Probe Gas Chromatogragh Mass Spectrometer Experiment Results and Lessons learned"*
10:30-11:10	**(Invited) T.R. Spilker and A. Coustenis** *"Mission Concepts for Titan Exploration"*
11:10-11:25	**R. Lorenz** *"Practical Balloon Designs for Titan"*
11:25-11:40	**A. Elfes, J. L. Hall, J. A. Cutts** *"Autonomy technologies for self-propelled and wind-driven aerobots"*
11:40-12:00	**Q&A and DISCUSSION**

12:00 – 13:30	**Lunch (for participants to excursion to Saint-Emilion)**
13:30 – 19:00	**Excursion to Saint-Emilion** *(precise timing to be confirmed)*

FORMATION OF GIANT PLANETS AND THEIR ATMOSPHERES: ENTRY PROBES INTO SATURN AND BEYOND

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Comparative planetology of the outer solar system is key to the origin and evolution of the Solar System, and, by extension, extrasolar systems. In particular, elemental abundance, particularly those of the heavy elements (>^{4}He), is critical for constraining the formation models. It requires measurements in the well-mixed tropospheres. From in situ measurements, the Galileo probe mass spectrometer yielded the abundances of C, N, S (respectively from CH_4, NH_3 and H_2S), Ar, Kr, Xe, $^{14}N/^{15}N$, $^3He/^4He$, D/H, and isotopes of the heavy noble gases, but the oxygen elemental abundance (from H_2O) is unknown as the probe entered a meteorologically anomalous region – a five-micron hot spot. The Juno spacecraft is expected to measure the well-mixed water at Jupiter in 2016. While the abundance of critical elements will then be in hand for Jupiter, only the carbon abundance at Saturn is known from Cassini orbiter. The abundances of other elements are needed, as on Jupiter. It is only through comparison of Saturn with Jupiter that a reasonably good understanding of the formation of these gas giants and their atmospheres may be obtained. Entry probes capable of making measurements to 10 bars at Saturn, together with microwave radiometry on carrier spacecraft for determining deep water abundance, will be a powerful technique for obtaining the needed data. Complementary and contextual information such as the atmospheric dynamics and gravity will greatly enhance the value of such a mission. In the long run, comparison with the ice giants, Uranus and Neptune, will be important. Polar orbiters with probes at Neptune and Uranus should be in our plans for Flagship missions in the 2020's. Considering the complexity, cost and risk of the probe missions and the availability of immense scientific and technical expertise and resources world-wide, the only sensible way to accomplish such ambitious goals is for NASA, ESA and the Asian nations to collaborate. In that spirit, a probe mission to Saturn is being developed for the near-term. A Cosmic Vision proposal to ESA, Kronos, is being prepared for Europe to contribute probes to a potential NASA New Frontiers mission where the US will be responsible for the carrier spacecraft and microwave radiometry. Payload instruments will be developed jointly. This talk focuses on the science and the need for probe missions. Other talks in the session will discuss Kronos and mission architecture studies. <http://umich.edu/~atreya>

KRONOS - SATURN EXPLORATION WITH PROBES

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And the KRONOS team

To understand the origin of the Solar System we must understand the origin of its two largest planets, Jupiter and Saturn. Thanks to the Galileo mission with its atmospheric probe and the Juno mission, we will have by 2016 fundamental information about Jupiter's composition, interior structure, magnetic field and atmospheric dynamics. We now need to achieve the same level of understanding about Saturn, which requires in situ measurements.

We therefore propose Kronos, a mission consisting of a solar-powered carrier to be built by NASA and two identical probes to be built by ESA which would be sent into Saturn's atmosphere. Kronos would use results and heritage from the Galileo, Cassini-Huygens and Juno missions. It will therefore allow the crucial comparison between measurements made in Jupiter and in Saturn that we need.

The probe measurements of elemental and isotopic composition are key to the studies of the formation and evolution of the Solar System. The mission will further allow better understanding planetary interiors, planetary atmospheric dynamics and their meteorology, as well as the generation of magnetic fields. It will improve our models of the origin and evolution of giant planets in general, including extrasolar planets.

Kronos would be a joint NASA/ESA mission. We will descibe the proposal submitted to the ESA Cosmic Vision 2015-2025 programme and in particular the probe aspects.

Mission Design Consequences of a Jovian Atmospheric Probe

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EADS Astrium has recently completed the second of two Technology Reference Studies for ESA on the subject of missions to Jupiter - the Jupiter MiniSat Orbiter [JMO], and the Jovian System Explorer [JSE]. The JSE study involved the pre-phase A design of a dual spin-stabilised spacecraft mission (with primarily magnetospheric science objectives), with the interesting addition of including an entry probe destined for the atmosphere of Jupiter.

The mission design performed during the course of the JSE study enabled the consequences of including an atmospheric probe into a Jupiter mission to be assessed and more clearly understood. As could be expected, the consequences of aeroprobe inclusion are considerable, with the presence of the probe impacting all aspects of the mission, from spacecraft configuration through to operations and mission analysis.

This paper will present a thumbnail design of a possible Jovian atmospheric probe, based on a recent study by the ESTEC CDF, with some brief discussion of possible design improvements and a general discussion of the design challenges. The impact of this probe on the overall JSE mission design will then be discussed - in particular the impact of the probe on the configuration, operations, mass and ultimately feasibility of the mission. By considering the design of the JMO TRS, some consideration is then given to the likely effect of an aeroprobe on Jupiter mission architectures in general.

It is demonstrated that the decision to include an aeroprobe within a future Jupiter mission will have arguably overbearing consequences for the entire mission architecture. This is particularly the case for spin-stabilised missions where the requirement to accommodate the probe on the spin-axis has very important ramifications. The lesson for the aeroprobe community is clear: in order to make a Jovian atmospheric probe attractive for inclusion in a future ESA mission, it must be carefully considered within a suitable overall mission context.

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5th International Planetary Probe Conference/Workshop
June 23 – 29 June, 2007, Bordeaux, France

Aerothermal Analysis for Planetary Entry Probes Using DOTNET Framework and OLAP Cubes database

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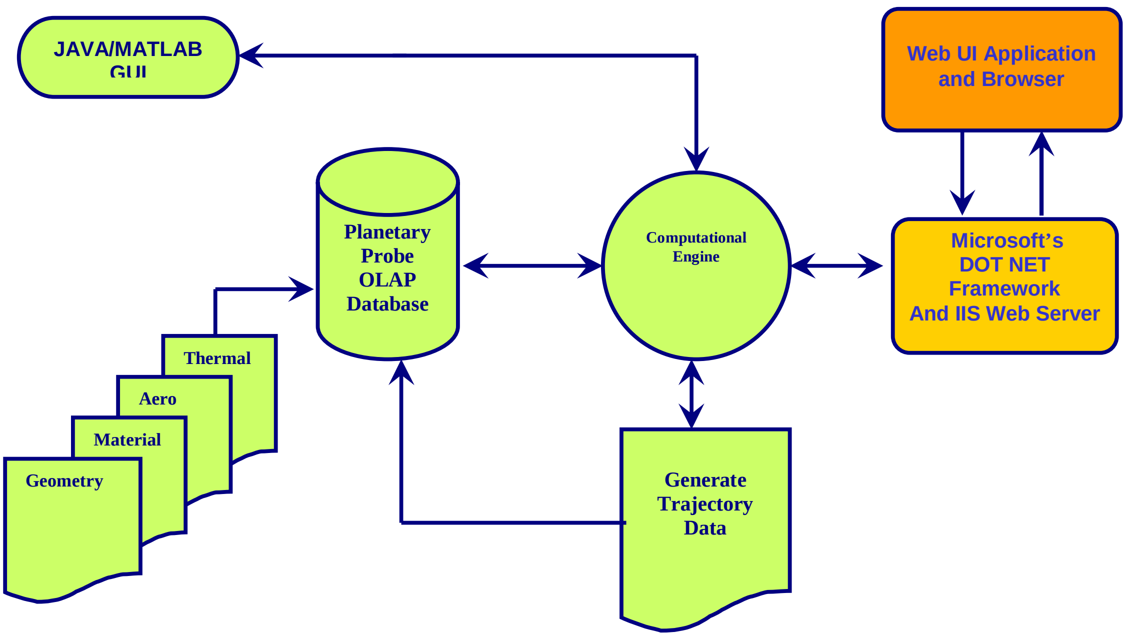
Abstract

This publication presents the architecture integration and implementation of various modules in *Sparta* framework. *Sparta* is a trajectory engine that is hooked to an OLAP database for Multi-dimensional analysis capability. OLAP is an Online Analytical Processing database that has a comprehensive list of atmospheric entry probes and their vehicle dimensions, trajectory data, aero-thermal data and material properties like Carbon, Silicon and Carbon-Phenolic based Ablators. OLAP has the capability to run in one simulation several different trajectory conditions and the output is stored back into the database and the data can be queried for appropriate trajectory type. For instance, given any one planetary probe dimensions from the database, an OLAP simulation can be setup to run for three types of trajectory: *Nominal*, *Undershoot and Overshoot trajectory*. This aids as a very good preliminary design tool for a trajectory based TPS sizing and design. The Trajectory code is written in pure Java and parallelized using Java threads that are internal to the Java. The graphical user interface (GUI) for the trajectory engine is coded in Java and Matlab since Java and Matlab are interoperable. A GUI driven web application is coded in .NET and deployed in the web for the trajectory engine. .NET Framework acts as a middleware layer between the trajectory engine and the GUI and also between the Web GUI and the OLAP database. Thus the trajectory can be run as a standalone engine and also from the web. Trajectory Reconstruction module using *Kalman filters* for the descent mode for planetary probes is implemented as an optional module in *Sparta* and discussed in this paper. *Sparta* provides many hooks to CFD solvers. A GUI in Java is coded to generate input decks for the industry standard NASA DPLR code. Thus a pure trajectory driven CFD capability is demonstrated in this publication.



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The Huygens Probe Gas Chromatograph Mass Spectrometer experiment, results and lessons learned.

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Abstract

The Gas Chromatograph Mass Spectrometer (GCMS) experiment was part of the instrument complement on the Huygens Probe to measure *in situ* the chemical composition of the atmosphere the Saturnian moon Titan during the probe descent and to support the Aerosol Collector Pyrolyser (ACP) experiment by serving as detector for the pyrolization products. The GCMS employed a quadrupole mass filter with a secondary electron multiplier detection system and a gas sampling system providing continuous direct atmospheric composition measurements and batch sampling through three gas chromatographic (GC) columns. The mass spectrometer employed five electron impact ion sources with available electron energies of either 70 or 25 eV. Three ion sources served as detectors for the GC columns and two were dedicated to direct atmosphere sampling and ACP gas sampling, respectively. The GCMS gas inlet was heated to prevent condensation, and served to evaporate surface constituents after impact.

The GCMS collected data from an altitude of 146 km to ground impact for a time interval of 2 hours and 27 minutes. The Probe and the GCMS survived impact and collected data for 1 hour and 9 minutes on the surface until signal loss by the orbiter. 5634 mass spectra were collected during descent and 2692 spectra on the ground over a range of m/z from 2 to 141. Eight gas chromatograph samples were taken during the descent and two on the ground.

The major constituents of the lower atmosphere were confirmed to be N_2 and CH_4. The methane mole fraction was fairly uniform in the stratosphere. The mole fraction began to increase below the tropopause, at about 32 km altitude, monotonically toward the surface, reaching a plateau at about 8 km at a level near saturation.

A steep increase of the methane signal was observed immediately after surface impact, suggesting evaporation of surface condensed methane due to heating of the surface in the vicinity of the inlet by the GCMS sample inlet heater.
Volume mixing ratios of Argon 40 and Argon 36were determined. The other primordial noble gases were below the 10^{-8} detection threshold.
The isotope ratios of $^{12}C/^{13}C$, $^{14}N/^{15}N$ and D/H were determined from measurements of methane, molecular nitrogen and hydrogen respectively.
Outgassing from the surface was also observed of carbon dioxide, ethane, cyanogen and other more complex hydrocarbons and nitriles.
The unexpected long survival of the probe on the surface suggests that a very broad in scope design should be considered for future experiments.

MISSION CONCEPTS FOR TITAN EXPLORATION

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Despite the fact that Titan, compared to Earth, is completely alien in terms of temperatures and surface composition, Titan appears to be the most Earthlike body in the solar system in terms of the balance of evolutionary forces: volcanism, impacts, tectonics, erosion by wind and liquids, etc. This, and its potential for preserving products of prebiotic chemistry long ago lost at Earth, make it a destination rich in phenomena of great scientific interest. These phenomena extend from the far upper reaches of its thick atmosphere to its interior. In the upper atmosphere, simple atmospheric constituents and energetic photons and particles collaborate to create more complex chemistry, including complex organic compounds that fall toward the surface. Farther down the chemical products condense and form a series of haze layers seen by Cassini imaging instruments, and still farther down they form hydrocarbon clouds and dense hazes that obscure the surface at all but a narrow set of infrared wavelengths. These clouds and hazes ride planetary winds associated with global and local weather, including convective activity, precipitation, and possibly lightning. At the surface the descending products accumulate as solids or liquids on bedrock that appears to be mostly water ice. The winds and precipitants sculpt that bedrock into Earthlike landforms, and produce other Earthlike landforms such as dunes and lakes. The surface also could act as a site for heterogeneous and/or catalytic chemistry, especially in the presence of energy or primordial reactants emerging from the interior. This abundance of phenomena at Titan make it an ideal destination for investigations based on a variety of atmospheric and surface platforms, from simple entry probes to landers, lighter-than-air and even heavier-than-air aircraft, boats, submarines – all have features attractive to the science community.

The planetary exploration community is busily assessing all these exploration options. Recent studies, and others currently underway, are examining the range of high-priority science objectives to be pursued at Titan, and are weighing each of the potential mission architectures to find those that yield the best science for the mission resources available. Attaining science value that justifies a mission is not easy: data from the Cassini/Huygens mission has allowed spectacular progress in understanding Titan, so a follow-on mission must achieve even better results. Missions with flight system elements that perform in situ science after entering Titan's atmosphere generally score well in that regard. Communications issues are important considerations. Recent studies indicate that an in situ element such as an aerobot or lander can return 50 to 100 times as much data by relaying to a Titan orbiter compared to direct-to-Earth downlink only.

This presentation will cover in detail the science objectives to be pursued via atmospheric entry platforms and accompanying elements such a Titan orbiter, and will discuss architecture options for such missions and recent studies.

PRACTICAL BALLOON DESIGNS FOR TITAN.

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NASA has commissioned four short studies of possible Flagship-class (~$3B) missions to the outer solar system: to Europa, the Jovian System, Enceladus and Titan. The latter destination is showing itself in Cassini data to be an extraordinarily rich target scientifically. Further, by virtue of its thick atmosphere, it is the easiest target for delivery of instrumentation to the surface and near-surface.

The study is in its early stages, and scientific priorities are still being evaluated. It is already clear, however, that Titan is a complex integrated system that needs exploration at all scales, and that orbiters, balloons and landers all address different facets of the system. In contrast to previous studies where a complex balloon or airship aimed to address surface science goals, we are exploring architectures with separate lander and balloon.

In this architecture, the balloon (possibly an 80kg 'hot-air' balloon lofted by the >1kW of 'waste' heat from an RTG) is a much smaller and simpler vehicle, engaging a much lower degree of technical risk. Its science return could nonetheless be formidable, however, circumnavigating Titan at least once in a one year mission and returning (for example) high-resolution surface images, spectra and subsurface radar.

Helium balloons of this class are possible, but appear less mass-efficient, and would not allow easy altitude control, and would be potentially life-limited by leaks. However, purely passive radar-reflective balloons for wind tracking, or any effort to attain high altitudes on Titan (e.g. ~50km is quite possible), would be best done with a helium balloon. The various options are discussed, and previous balloon and airship concepts are reviewed. The Titan mission study is due for completion in August 2007 : NASA will thereafter consider which (if any) flagship mission options to pursue further.

AUTONOMY TECHNOLOGIES FOR SELF-PROPELLED AND WIND-DRIVEN AEROBOTS

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Introduction: Two of the three top priority flagship missions proposed in the NASA 2006 Solar System Exploration Roadmap [NASA06] are in situ missions to Titan and Venus using lighter-than-air vehicles (the Titan Explorer and Venus Mobile Explorer missions, respectively). The dense atmospheres at Titan and Venus enable the use of buoyant robotic vehicles (aerobots) that can be either self-propelled (airships) or wind-driven (balloons). Aerobots can provide extensive geographical coverage over multi-month time scales with minimal power consumption [Cutts04, Hall02, Hall04, Elfes04]. Airships provide additional scientific value by being able to fly to specific science targets, while balloons are simpler in their design. In this paper, we outline the challenges involved in autonomous aerobot exploration of Titan, the required autonomy capabilities, and the aerobot autonomy architecture that the authors are developing at JPL.

Challenges: The main challenges for aerobot exploration of Titan include: large communication latencies, with a round trip light time of approximately 2.6 hours; extended communication blackout periods with a duration of up to 9 Earth days, caused by the rotation of Titan and its orbital occlusion by Saturn; extended mission duration, on the order of six months to one year; and operation in a substantially unknown environment, with very little data on wind patterns and meteorological conditions, and only low-level resolution maps of surface topography. This will require Titan exploration vehicles to have autonomy capabilities that go substantially beyond what has been demonstrated for Mars rovers.

Autonomy Architecture: The core autonomy areas for a Titan aerobot mission include 1) vehicle guidance and flight control, 2) navigation planning and execution, 3) localization of the aerobot on the surface of Titan, 4) multi-modal mapping, 5) autonomous science, 6) vehicle resource management and health/safety monitoring, and 7) integrated mission planning and execution. We will discuss these areas, and assess to what extent the autonomy requirements for self-propelled and wind-driven aerobots are either similar or substantially different. We will also present our progress in the development of the core autonomy technologies required for Titan aerobots, and show results from both simulation studies and autonomy tests conducted in the Mojave desert with an 11 m self-powered aerobot.

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SESSION VI: Entry, Descent, and Landing Technologies for Planetary Missions (N. Cheatwood, D. Lebleu) Thursday, June 26, 8:30-12:00

8:30-8:55	**(Invited) J.-M. Muylaert, et al.** *"Expert Aerothermodynamic Flight Instrumentation Environment andIintegration"*
8:55-9:15	**J.E. Theisinger** *"Hypersonic Aerodynamic Aeroshell Shape Optimization"*
9:15-9:35	**A.A. Dyakonov, et al.** *"Aerodynamic Interference Effects and Aeroheating Augmentation due to Operation of the Reaction Control System during Atmospheric Entry"*
9:35-9:55	**R.A. Gowen, et al.** *"Kinetic Micro-Penetrators for Exploration of Solar System Bodies"*
9:55-10:05	**Break**
10:05-10:30	**(Invited) M. Adler** *"Mars EDL Capability Drivers and Technology Needs"*
10:30-10:50	**I. Clark, et al.** *"Design and Development of an Inflatable Supersonic Tension Cone Decelerator"*
10:50-11:10	**B. Laub, et al.** *"Requirements for Development of Thermal Protection Materials for Multiple Missions"*
11:10-11:30	**S.T. Surzhikov, D.V. Kotov** *"Computational Aerophysics of Space Vehicles for Planetary Missions"*
11:30-12:00	**Open Mic, Q&A, Panel Discussion, etc**
12:00-13:30	**Lunch**

Thursday Lunch offered by :

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EXPERT AEROTHERMODYNAMIC FLIGHT INSTRUMENTATION ENVIRONMENT AND INTEGRATION

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The paper reports on the aerothermodynamic (ATD) environment of the EXPERT configuration associated with the 5km/sec suborbital flight. A status report is given on the flight measurement technique developments and qualification with emphasis on the thermal protection system (TPS) integration issues. Special attention is given to the design of the flight measurement sensors themselves, their integration into the TPS as well as to the measurement of the free stream parameters during re-entry using an Air Data System (ADS).

The paper will address the numerical design work to optimise the location of sensors in order to enhance the phenomena of interest, such as; definition of nose radius so as to avoid contamination of the boundary layer due to passive/active oxidation of the nose thermal protection system, definition of the admissible step between nose and conical parts to avoid premature boundary layer transition, nonequilibrium maximum laminar heating on metallic PM1000 parts of the vehicle including effects of catalytic recombination validated through recent experiments in plasmatron, turbulent reattachment of separated boundary layer on the flaps and in the vicinity of the corners of these flaps.
e.g. the paper will address how the mach number was selected as to trigger transition with the roughness elements and how the design was made taking into account the estimated overheating induced by the elements themselves.
Finally an update will be given on the flight schedule.

HYPERSONIC AERODYNAMIC AEROSHELL SHAPE OPTIMIZATION

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For entry descent, and landing systems, the performance of the aeroshell during hypersonic flight is of critical importance. For a given unpowered trajectory, two vehicle parameters dictate this performance – the ballistic coefficient and lift-to-drag ratio. The ballistic coefficient is the ratio of the vehicle's mass to the drag-area produced by the aeroshell, thus quantifying the ability of the vehicle to decelerate prior to terminal descent and landing events. Increasing an aeroshell's drag-area decreases the ballistic coefficient of the vehicle and improves its capability to decererlerate efficiently and safely – hence the motivation for large inflatable aerodynamic decelerators. For ballistic vehicles, reducing the ballistic coefficient is the complete story. However, adding lift to an entry system can be advantageous or even necessary for certain missions. This lifting capability is quantified by the lift-to-drag ratio. In achieving a certain lift-to-drag ratio, care must be taken not to degrade the vehicle's ability to decelerate through large reductions in drag-area that then increase ballistic coefficient. There is therefore a significant trade between ballistic coefficient and lift-to-drag ratio, with drag-area being the common denominator. Thus, an optimum entry system aeroshell should achieve a specified lift-to-drag ratio in conjunction with a maximum amount of drag-area.

This work develops the capability to identify hypersonic aeroshell shapes that can achieve a specified lift-to-drag ratio and a maximum amount of drag-area through several different approaches to shape design, each with an increasing level of generality. The most basic approach manipulates standard parameters associated with analytic aeroshell shapes like the sphere-cone and ellipsled. A more general approach produces surfaces of revolution after manipulating the control points of a spline profile. The most general approach then manipulates the control points of a spline surface, thus allowing for the creation of non-axisymmetric aeroshell shapes. The parametric polynomial formulations of the Bezier and B-Spline curves and surfaces have been employed due to their attractive properties in shape design.

Hypersonic aerodynamic analyses are carried out using Newtonian flow theory panel methods in MATLAB. These analyses are rapid and ideal for integration into an optimization environment. This environment is created in ModelCenter for each approach to shape design with the option of a variety of optimization methods. In addition to the lift-to-drag ratio constraint, size constraints are imposed on the aeroshell, as determined by minimum payload volume requirements and launch vehicle shroud size restrictions. Note that while these aeroshell shapes are optimal from a hypersonic aerodynamics perspective, stability and aerothermodynamic considerations must be included for complete entry system optimality. This study therefore serves as an initial step in identifying ideal aeroshell shapes, from which the designer can identify the aerodynamic sacrifices required to satisfy stability and aerothermodynamic constraints. Test cases include a large-mass robotic-class Mars mission and a human-class Mars mission with comparisons made to traditionally-considered aeroshell shapes.

AERODYNAMIC INTERFERENCE EFFECTS AND AEROHEATING AUGMENTATION DUE TO THE OPERATION OF THE REACTION CONTROL SYSTEM DURING ATMOSPHERIC ENTRY

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Introduction: The next generation of Mars exploration landers must precisely deliver scientific payloads to sites of interest, unlike previous Mars missions. The past missions, such as Viking and Pathfinder, performed landings to within 100s of kilometers from their targets using an unguided atmospheric entry. Guided entry of a capsule with a relatively high lift-to-drag ratio will allow landing to within 10s of kilometers from the target with a significantly more massive payload. Successful guided entry requires the use of a reaction control system (RCS) for both attitude correction and entry guidance maneuvers. Various aspects of the entry, descent and landing (EDL) system performance may be impacted by the operation of the RCS during entry. This paper illustrates the risks that arise from the gasdynamic interaction of the entry vehicle (EV) and RCS, and which require attention in the areas of aerodynamics and control, and aerothermal environments. This paper will review the methods to address the design challenges associated with integration of RCS into the atmospheric entry system. Among these challenges is the analysis of the potential for the aerodynamic interference due to both the direct jet plume impingement and more complex plume interactions with the wake flow. These interactions can result in enhanced aeroheating, requiring that a different approach to the thermal protection system (TPS) selection and sizing be used. The recent findings for Mars Science Laboratory and Mars Phoenix will be presented to help illustrate some of the phenomena. Current design solutions will be discussed.

KINETIC MICRO-PENETRATORS FOR EXPLORATION OF SOLAR SYSTEM BODIES.

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Abstract: We report on the potential usefulness of kinetic micro-penetrators for the exploration of various types of Solar System bodies, including planets their satellites, and near Earth objects. We consider their potential scientific return, resource requirements and costs compared with soft landers, and outline their heritage and current state of development. Following the recent round of proposals for the European Space Agency Cosmic Visions, penetrators have been put foreward for three missions: the moon, Europa and Titan. Each situation involves an number unique challenges which will be addressed in the presentation. The paper also includes technology roadmaps for various penetrator science instruments and key penetrator technologies (such as batteries and RHUs).

Mars EDL Capability Drivers and Technology Needs

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NASA plans, desires, and dreams for future robotic Mars exploration will place increasing demands on entry, descent, and landing systems. These in turn require technology development and maturation prior to the start of projects, whose success depends on advanced technologies. Drivers arise from the desire to land higher-mass vehicles, achieve more accurate landings, as well as demands in an entirely different direction for smaller, lower cost landers to provide broad surface coverage. The technology developments cover many diverse areas including thermal protection systems and modeling, drag devices, impact devices and survival, propulsion systems, as well as terminal descent sensors and algorithms. This talk will describe the mission drivers and the technology developments needed to enable future robotic Mars missions.

Design and Development of an Inflatable Supersonic Tension Cone Decelerator

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Planetary entry systems have historically relied on aerodynamic drag for deceleration. However, the technology that generates this drag, namely rigid aeroshells and supersonic and subsonic parachutes, is quickly reaching its limitations. Inflatable aerodynamic decelerators (IADs) represent a potential technology path that can relax the size and deployment limitations of aeroshells and parachutes. That is, an IAD can serve to increase the drag area provided by a rigid aeroshell while still keeping the diameter of the aeroshell within launch vehicle payload fairing limits. Additionally, the aerodynamic characteristics of IADs allow them to be deployed at higher Mach numbers and dynamic pressures than can be achieved by current supersonic parachute technology. This combination of earlier deployment and increased drag over the rigid aeroshell allows for the landing of heavier payloads. This paper reviews historical efforts aimed at developing supersonic IADs and describes an analysis and test program designed to increase the technical maturity of an inflatable supersonic tension cone decelerator.

A significant amount of work was performed in the late 60's and early 70's on maturing the IAD concept. Initial efforts in the conceptual domain explored the advantages provided by both attached and trailing supersonic IADs. This work included performance comparisons to both subsonic and supersonic parachutes and outlined flight regimes (Mach and dynamic pressure) in which IADs were calculated to be lighter than parachutes. Subsequent IAD development efforts sought to analyze performance under simulated supersonic flight conditions. One particularly well-studied configuration was a ram-air inflated design whose shape was tailored to provide uniform tensile stress throughout the entire fabric. Several dozen small scale isotensoid models (1.5 m in diameter) were tested for aerodynamics, stability, and inflation behavior at Mach numbers up to 4.5. Results of those tests demonstrated an excellent ability to accurately model aerodynamic performance, fabric stress, and deployment times.

Another configuration studied concurrently with the isotensoid shape was the tension cone. The tension cone concept consists of a flexible shell that is uniquely shaped so as to remain under tension and thus resist shape deformation. The shape itself is analytically derived on the basis of a predefined pressure distribution, a desired drag contribution of the tension shell, and an assumed constant ratio of circumferential to meridional stress. The tension in the shell is resisted at one end by a rigid forebody and at the other end by a compression ring, which typically consists of an inflated torus.

When compared to alternative configurations, the tension cone is promising in that it allows for a reduction of material acreage and thus, potentially, mass. However, the tension cone concept is not without its own design issues. These include preventing flow separation along the forebody, preventing buckling of the inflated torus, and preventing aeroelastic oscillations. Previous wind tunnel tests of rigid tension cone models have provided guidelines for preventing flow separation. Additionally, a large body of work has been developed for predicting the buckling behavior of inflated toroids. The combination of previous rigid model wind tunnel tests and tension cone structural theory allow for a detailed design analysis that provides insight into the tension cone design space. In particular, trends of drag performance, inflation gas mass, and material requirements are attained as a function of configuration dimensions and dynamic pressures.

Although a theoretical basis exists for designing and estimating several aspects of tension cone performance, much of this theory has yet to be validated. Furthermore, theoretical models for accurately predicting aeroelastic tension cone behavior do not presently exist. To address these issues and to further advance the maturity of the tension cone concept an effort has begun to explore the behavior of an inflatable tension cone in a supersonic wind tunnel. Currently planned for November of 2007, the goal of this testing is to acquire data to characterize the behavior of a flexible tension cone decelerator under simulated flight conditions.

Requirements for Development of Thermal Protection Materials for Multiple Missions

by
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Abstract

In over 45 years of U.S. robotic and crewed missions involving atmospheric entry, only a handful of thermal protection materials were developed for specific missions. Examples include Avcoat 5026-39/HC-G for Apollo, SLA-561V for Mars Viking, and Reusable Carbon-Carbon (RCC) and LI-2200 tiles for shuttle. The majority of missions employed existing materials, often with selection based on trade studies or screening tests among competing candidate materials.

Unfortunately, it is also quite common that new missions use a “heritage” argument to baseline an existing TPS system without having the comprehensive knowledge of either the limitation of the material performance or the capabilities required of the TPS for the newer missions. Often not knowing the performance limitations of existing materials and systems in an entry environment more severe, or at minimum, very different than prior applications has led to either accepting the risk, making a valiant effort to understand the performance prior to launch, or opting to design conservatively to account for the risk.

Since the missions have been infrequent and projects have been constrained to focus on their immediate need there is typically little, if any, performance data at the more severe conditions. Consequently, the project for the new mission is burdened with developing the requisite database, design models and reliability estimates applicable to their mission. Examples include PICA, successfully used on Stardust and currently a candidate for Orion’s CEV, and SLA-561V previously used successfully on Mars Viking, Mars Pathfinder and Mars Exploration Rover and currently the selected TPS for Mars Science Laboratory.

Often, in extending the database for a selected TPS material, projects expend significant resources only to discover unanticipated material performance characteristics that suggest that their material selection may not have been the best choice. In those instances projects are very reluctant to change materials given their investment of resources and typical schedule constraints.

NASA has already done the *easy* missions. It is generally accepted that thermal protection requirements for future robotic missions will be more demanding. Currently, missions to the Gas Giants are constrained by the performance limitations of existing materials and the lack of adequate ground test facilities to qualify new materials. What’s needed is a new *philosophy* wherein material developers and/or their sponsors conduct a comprehensive set of experiments to *fully characterize and model material performance over a broad range of conditions*, i.e., to explore the performance limits of a TPS material and system. If such a database existed for a set of materials covering a broad range of density, chemical composition and architecture, projects

could select materials with the knowledge that they were capable of satisfying their mission parameters.

The paper will discuss the challenges in qualifying existing TPS materials for CEV and MSL applications and will provide an outline of the test and analysis requirements to fully qualify a candidate TPS material. New missions are being planned to destinations that we have visited before such as Venus and Titan and new destinations such as Saturn and Neptune in decades to come. Will we be ready? What should we, the TPS technologists, do now? This is the quest & the question we plan to answer.

COMPUTATIONAL AEROPHYSICS OF SPACE VEHICLES FOR PLANETARY MISSIONS

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The problem of creation of reliable engineering codes for prediction aerothermodynamics of space vehicles for planetary missions, and particularly for Martian missions, is one of the top-priority problems for modern aerophysics. An effective way to create such reliable models and codes is the development of engineering codes based on different physical-chemical models, and comparison numerous calculation data between itself and with available experimental data. One of such models, the computational radiative gas dynamic model NERAT (Non-Equilibrium Radiative Aerothermodynamics), which is intended for prediction of aerothermodynamics of space vehicles entering into planetary atmosphere, is presented. The model is based on equations of viscous, heat conducting, chemically non-equilibrium radiating gas, and includes databases of thermodynamic, kinetic and radiation properties, which are sufficient for prediction of flow field parameters around space vehicles entering into planetary atmosphere. A distinguishing feature of the code presented is the unification of gasdynamic, kinetic and radiation codes into united computing system. One more significant feature of the model is prediction of translational temperature through the Fourier – Kirchhoff equation. An effectiveness of such approach is demonstrated in cases under consideration.

Numerical simulation results for radiative heating of space vehicle Mars Sampler Return Orbiter (MSRO) are presented and analyzed.

The calculations were performed by the radiative gas dynamic code NERAT for four kinetic models of non-equilibrium chemical reactions in gas phase. The first one is the Park kinetic model [1,2]. The second model contains kinetic model formed on the base of analysis of recently published data. Vibrational excitation is taken into account in both kinetic models.

Prediction of radiative heating of MSRO surface was performed for four trajectory points and for two assumptions concerning the space vehicle wall catalytic effects. In the first case the surface was presumed as non-catalytic, and the pseudo-catalytic surface with 97% CO_2 and 3% N_2 at the wall was presumed for the second case.

Numerical simulation results are grouped for studied trajectory points. Each group contains full data on convective and radiative heating, and it is divided in its turn on two sub-groups for pseudo-catalytic and non-catalytic surface. Each group of presented calculation data contains temperature field and distribution of CO_2 mass fraction. Total radiation flux and convective flux on MSRO surface is shown through whole surface, and some figures show spectral radiation fluxes at five points on the surface.

Presented data show that differences between total radiation fluxes for catalytic and non-catalytic walls can be observed only for high-velocity points of trajectory were carbon dioxide is decomposed behind shock waves. It is obvious that just in this case catalytic property of a space vehicle wall plays significant role in association of C and O atoms, and oxidation of CO molecules up to carbon dioxide. As for the wall catalytic surface, one can observe decreasing of total radiation flux at the front surface of space vehicle in comparison with non-catalytic case.

Strategy of on-going efforts on validation of the computing code and validation of different physical-chemical models are presented.

The concept developed in the study can be applied to different configurations of space vehicles and different atmospheres, as it includes computational modules for prediction of thermodynamic, transport and spectral optical properties for various gas species.

Some of used rates of chemical kinetic processes are predicted by ab-initio approach based on molecular dynamic methods.

So, the presented concept of aerothermodynamics of space vehicles is based on multi-level approach, which includes ab-initio description for prediction kinetic and radiative properties and computational fluid dynamic models for prediction aerodynamics and heating of space vehicles.

References

1. Proc. of the International Workshop on Radiation of High Tempereture Gases in Atmospheric Entry. European Space Agency, SP-533, Dec. 2003, 200 p.
2. Workshop 2003 – “Radiation of High Temperature Gas,” TC3: Definition of an axially symmetric test case for high temperature gas radiation prediction in Mars atmosphere entry,” NG104-07-TF-001-CNES, 2003.

SESSION VII: Emerging, Enabling, and Extreme Environment Technologies; Cross-Cutting Technologies
(L. Peltz)
Thursday, June 28, 13:30-17:00

13:30-14:00	**(Invited) A. Keys** *"Radiation Hardened Electronics for Space Environments"*
14:00-14:15	**P. Estabrook, et al.** *"Robust, Miniaturized Avionics Suite for Network-Enabled Deep Space Probes"*
14:15-14:30	**W. Schmidt, A Mälkki,** *"Single-Chip FPGA-based Processor-Sensor-Controller"*
14:30-14:45	**L. Peltz, et al.** *"Silicon Germanium (SiGe) Technology for Europa Missions"*
14:45-15:00	**M. Alles, et al.** *"Simulation of Radiation Effects for Performance Prediction and Design Optimization of Electronic Systems"*
15:00-15:15	**Break**
15:15-15:45	**(Invited) W. Johnson** *"Electronics Packaging for Extreme Environments"*
15:45-16:00	**G.W. Hunter, et al.** *"High Temperature Electronics, Communications, and Supporting Technologies for Venus Missions"*
16:00-16:15	**J.P. Roux, et al.** *"PU-238 Radio-Isotopic Thermoelectric Generator (RTG) for Planets Exploration"*
16:15-16:30	**D. Ila, et al.** *"Highly Efficient Thermoelectric Materials: Nano-Layered Nanoclusters"*
16:30-16:45	**B. Farkin, et al.** *"Real-Time 3D Collaborative Design Simulation in Support of NASA and ESA Planetary Exploration Programmes"*
16:45-17:00	**Q&A and technical summary**

“Radiation Hardened Electronics for Space Environments (RHESE) Program”
Invited Talk for Session VII

By Michael Watson

Technology innovation for Radiation Hardened Electronics for extreme environments in Space exploration.

Full abstract to follow.

Robust, Miniaturized, Avionics Suite for Network-Enabled Deep Space Probes

Authors: Polly Estabrook, Loren Clare, Norm Lay, and Jackson Pang

Abstract
(Not Cleared for Publication)

Below is a brief description to be replaced by Abstract once it is cleared.

This paper will discuss a miniaturized transceiver design with a highly efficient network layer protocol capable of being implemented in a FPGA. The resultant communication system can be integrated with avionics in order to create the core of a deep space probe. The paper will describe potential applications for these small probes. The microtransceiver design and performance will be detailed and the implementation of the Disruption Tolerant Networking (DTN) protocol will be discussed.

SINGLE-CHIP FPGA-BASED PROCESSOR-SENSOR-CONTROLLER

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Background: For the ESA SMART-1 mission we provided a plasma instrument which was designed to monitor the electron- and ion-density around the spacecraft in very different space environments: inside the Earth radiation belt, inside the Magnetosphere and around the moon either with the electric propulsion system active or passive. Besides needing a large dynamic range and high resolution the instrument had to work under extreme radiation conditions and at times with only little power available.

Implementation: The power constraints and radiation hardness requirements were met by eliminating as many analog circuits as possible. Space-qualified Digital-to-Analog- and Analog-to-Digital-Converters usually consume tens of Milliamperes, adequate processors much more, and need a significant mass for radiation shielding. For the SPEDE instrument (Spacecraft Potential, Electron and Dust Experiment) another concept was implemented. The processor function was implemented on a radiation tolerant FPGA (Field Programmable Gate Array) as a 16 MHz 16-bit RISC processor. A real-time sequencer for measurement control was implemented on the same circuit. Its control and status information was directly mapped into the processor registers saving interrupt handling and Input/Output functions.

A hardware memory management system inside the FPGA supported memory paging, separation of data and program area and the possibility of several different software versions residing in parallel in the attached radiation-hardened EEPROM. In case of a program failure e.g. caused by radiation, the FPGA-resident bootloader changed automatically to another software copy.

Analog voltages were generated under program control via an external resistor network, followed by a low-power radiation hardened operational amplifier. Forcing different FPGA-pin combinations to logical '0' , a linear 8-bit DAC-function was simulated. Analog signals were measured via Voltage-to-Frequency converters for up to 200kHz. The resulting pulses were then either measured in an adjustable time window defining the used integration time, or the length of a set of pulses was compared with the processor clock. The latter method provides better resolution for small signals around the center frequency, while the former gives a large dynamic range. The instrument itself had a maximum power consumption of 200mW

Possible applications: In applications which are mass, volume and power critical or are intended for difficult environmental conditions like planetary landers, balloons etc, this approach looks promising. Mass, volume and power demands can be further reduced by combining the remaining analog functions with the central controller in a single-chip hybrid or ASIC solution.

Silicon-Germanium (SiGe) Technology For Europa Missions

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Abstract

A Europa mission has been designated by the Decadal Study as the highest priority flagship mission for the next decade, and is likely to be the next large mission to the outer planets after Cassini. A mission to Europa would likely include landers and/or surface probes released from an orbiter. A 90-day surface mission is desired; however, in order to have a lifetime longer than a few hours, the electronics would need to be designed to operate at the extreme cold surface temperatures and in the extreme radiation environment.

Europa has extreme environmental conditions, with a surface temperature varying from a low of 50 Kelvin up to 125 Kelvin, with a mean temperature of around 100 Kelvin. Europa also has an extreme radiation environment: 20 Mrad-H_2O per month at the surface. These are harsh and challenging environments for electronics circuitry. SiGe technology is particularly suited to meet these demands. The SiGe will operate over a wide range of temperatures: down to 43 Kelvin (-230 deg C) without "warm boxes", and up to 398 Kelvin (+125 deg C), in normal operation. The performance characteristics of SiGe devices vary gracefully over this extreme temperature range, with no evidence of abrupt "killer" phenomena. The structure of SiGe devices also confers a "free perk" -- multi-Mrad total dose hardness, with no intentional hardening.

Designing SiGe chips for these extreme environments, translated into device parameters, require new design rules. Device models in design tools (Cadence) must be extended, calibrated upon experimental data. Packaging design must account for the different thermal coefficients of die, casing and bonding materials, and the variations in physical properties of these materials over the temperature range.

SiGe is obtained by introducing a small (5%-15%) amount of germanium (Ge) in the silicon (Si) lattice. Si and Ge are "almost-but-not-perfect match", and the resulting lattice is stressed. After years of consistent research, semiconductor manufacturers (primarily IBM) have developed a reliable fabrication platform, compatible with standard Si integrated circuits fabrications, which offers low cost and high integration. The lattice stresses in SiGe bring desirable properties to SiGe devices. Higher operating speeds are possible -- SiGe technology is well accepted in high-speed communication circuits. And, important for our applications, SiGe devices can operate reliably in the extreme cold typical of Europa and the other icy satellites of the outer solar system, without carrier freeze-out.

This paper presents the radiation environments in orbit and at the surface of Europa. It presents design characteristics for Silicon-Germanium circuitry for operating low temperature (to 40 Kelvin) and high radiation environments, along with preliminary test results. Further, the paper proposes a radiation testing regime that would expose the electronics to an equivalent radiation dose as would be experienced during a 90-day mission at Europa. This testing regime includes sequential exposure to high energy electrons, protons, and heavy ions, involving

combined radiation dosage at cyclotron facilities, and exposure to Van Allen Belt radiation in geosynchronous transfer orbit.

Boeing is part of a highly innovative team funded by NASA, which is developing low cost SiGe Integrated Electronics for Extreme Environments. This university-NASA-industry team led by Dr. Cressler at Georgia Institute of Technology also includes Jet Propulsion Laboratory, Auburn University, University of Tennessee, Vanderbilt University, University of Maryland, BAE Systems, IBM, and Lynguent Inc.

To Be Submitted

International Planetary Probe 5 Workshop
Bordeaux, France
25-29 June, 2007

Simulation of Radiation Effects for Performance Prediction and Design Optimization of Electronic Systems

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Radiation can cause parametric degradation, persistent or permanent damage, and transient errors in electronic systems. The effects of radiation may also be dependent on temperature. Shielding and thermal control can reduce the impact of some types of radiation at the cost of payload weight. By leveraging existing knowledge and data on how radiation interacts with the materials used to construct electronics, simulations of the radiation response of devices and circuits can be used to predict system performance in radiation environments. A number of process and design approaches exist for radiation hardening; the impact of process and design changes can be evaluated using simulations in order to optimize electronics for operation in such environments. Further, there are many technology options for fabrication of electronic devices, particularly integrated circuit components. Simulation of radiation effects, combined with test data, can provide a basis for making technology choices for operation in target environments. Optimization of technologies and designs can relax or eliminate the need for environmental controls in some cases; elimination of the need for thermal control or shielding can reduce payload weight. Further, use of advanced technologies can increase functional capabilities and reduce power consumption. Examples include use of SiGe BiCMOS and advanced (≤ 90 nm) CMOS technologies in space electronics applications. Since little experience exists on operation of advanced and emerging technologies in such environments, simulations can be used to predict the impact of radiation and temperature, and to develop and evaluate mitigation strategies.

Advances in computing power allow simulations at multiple levels of hierarchy, from Monte-Carlo based simulation of the basic interactions of radiation at the atomic level, through 3-dimensional device simulations of radiation response, circuit responses, and on to system level performance. One challenge in such a simulation flow is the instantiation of the radiation response simulated with high fidelity at interaction and device levels into higher level models with efficiency to allow simulation of larger circuits and systems, while maintaining fidelity to provide accurate results. When combined with calibration test data, such simulations can be a potent tool for *a priori* prediction of how a device or system will perform in a remote radiation environment, and provide a powerful capability to optimize technologies and designs. The presentation will discuss approaches to such simulations and provide examples of present efforts.

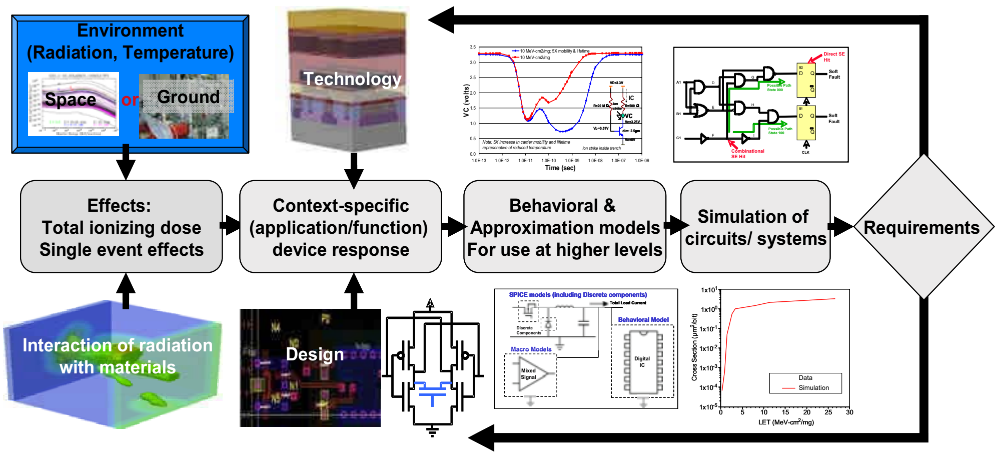

Fig 1. Example of simulation flow for radiation response.

"Electronics Packaging for Extreme Environments"
Invited Talk for Session VII

By Prof. R. Wayne Johnson

Technology innovation for packaging for electronics for cold environment (to 40 Kelvin) and to 500 deg C for 90-day Venus Lander mission.

Full abstract to follow.

High Temperature Electronics, Communications, and Supporting Technologies for Venus Missions

G. W. Hunter[(1)], P. G. Neudeck[(2)], G. M. Beheim[(3)], G. E. Ponchak [(4)], R. S. Okojie[(5)], M. Krasowski [(6)], and L.-Y. Chen[(7)]

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NASA Glenn Research Center (GRC) is presently leading the development of electronics and sensors capable of prolonged stable operation in harsh 500°C environments. These technologies are being developed for engine environments but also have planetary exploration applications. Given the previous lack of electronics that could process and transmit scientific data in Venus's 450°C lower-atmosphere, almost all proposed missions to explore this important planetary environment have been based on very limited duration (on the order of hours) of data collection and return. The ability of a spacecraft, including its electronics, to function and return useful data for far longer time periods (months) would significantly improve the scientific return gained from Venus surface missions.

For example, the recent emergence of wide bandgap semiconductors, including silicon carbide (SiC) and gallium nitride, has enabled short-term electrical device demonstrations at temperatures from 500°C to 650°C. Until recently however, these wide bandgap devices have demonstrated only limited (up to a few hours) of durability when electronically operated at these high temperatures. In order to support the needs of long-duration Venus surface missions, wide bandgap electronics technology must first demonstrate that it can achieve stable, long-term operation under electrical bias at 450°C temperature without significant changes in operating parameters.

NASA GRC is a world-leader in harsh environment electronics and sensor technology and is uniquely positioned to contribute to future Venus electronics systems. For example, NASA GRC has developed SiC-based transistor technology (including packaging) that has demonstrated continuous electrical operation of a 500°C low frequency AC voltage amplifier for over 450 hours [1] and other testing showing packaging durability at 500°C for 2000 hours. No other reported semiconductor transistor has demonstrated such continuous prolonged electrical operation in an ambient comparable to or exceeding Venus atmospheric temperature. In contrast to other proposed high temperature electronics approaches (such as miniature vacuum tubes), the NASA GRC SiC transistor technology is inherently compatible with integrated circuit manufacturing techniques, so that increasingly complex electronics could be implemented on a single SiC chip. Development of high temperature wireless communication based on SiC electronics has also been on-going at NASA GRC. This work has concentrated on the SiC electronics devices as well as the passive components such as resistors and capacitors needed to enable a high temperature wireless system. The overall approach is to make smart, integrated systems operable in harsh environments.

This paper discusses the development of SiC based electronics and wireless communications technology and its possible application in Venus missions. This electronics development includes the supporting technologies such as device contacts and packaging. Approaches to wireless communication in these environments will also be discussed. Further, characterization of Venus surface conditions also requires durable lightweight sensor technology which can operate in harsh environments. A brief overview of relevant sensor technologies and their compatibility with SiC based electronics will be given. It is concluded that the base technologies being developed for engine applications can have a significant effect on possible Venus missions.

1. L. Y. Chen, D. J. Spry, and P. G. Neudeck, "Demonstration of 500 C AC Amplifier Based on SiC MESFET and Ceramic Packaging," 2006 IMAPS International High Temperature Electronics Conference, Santa Fe, NM, May 15-18, 2006, pp. 240-246.

PU-238 RADIO-ISOTOPIC THERMOELECTRIC GENERATORS (RTG) FOR PLANETS EXPLORATION

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Nuclear power systems are a key enabling technology for space exploration missions:

- deep space missions to and beyond Jupiter, where the solar energy is to low,
- solar system planets exploration for powering equipment and appreciable prolongation of service life of landed equipment (rovers, autonomous stations etc.).

For these missions, future reliable and compact Radio-isotopic Thermoelectric Generators (RTGs) with electric power level from fractions of Watt up to several Watts seem to be the most probably required. The present paper presents different options for such generators.

"Angel" RHU and RTG have been developed and qualified for the "Mars-96" mission. One proposed option is to re-use the existing "Angel" RHU for these future RTGs. Results of computational and experimental approach show that RTGs based of compartmented cassette equipped with a few RHU "Angel" can reach near 1 W of electrical power.

For RTGs with an electric power from Watt up to several Watts, the development of new monoblock RHU is necessary with the realization of full development and qualification process.

Possible design options and achievable general parameters of RTG based on common RHS of 50 W_{heat} are presented. The electric power level is 4 W and 30 W, the designs are based on monoblock and cassette type RHU.

The assessments show that a RTG efficiency of about 8-10 % and a specific electric power of 3-5 W/kg are affordable, relying on existing techniques of fabrication of the cascade-type thermoelectric converters with an operation temperature up to 470 ^{0}C (TEC: BiTe+PbTe/GeTe) and up to 950 ^{0}C (TEC: PbTe/GeTe+SiGe).

Highly Efficient Thermoelectric Materials: Nano-layered Nanoclusters[1,2,3]

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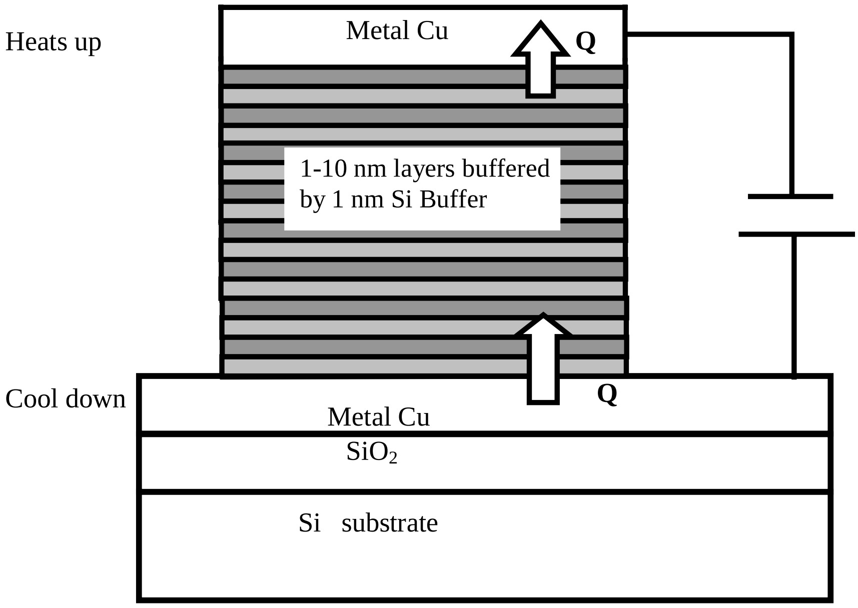
Abstract

The workers at the Center for Irradiation of Materials of Alabama A&M University (http://cim.aamu.edu/) have successfully produced highly efficient thermoelectric materials taking the advantage of interaction of nanolayers of nanoclusters (NLNC) of various material systems with each others, also known as quantum well quantum dot (QWQD). Theoretically the regimented quantum dot superlattice of any materials produces new physical properties such as new electrical band structure, mini-phonon bands, as well as improved mechanical and thermal properties. A proper choice of nanocrystals, host and buffer layer can result in a highly efficient thermoelectric generator (TEG) with efficiencies higher than 30% corresponding to figure of merit above 4.0. Such system, alone, can be used to substitute all present energy conversion units as well as the monolithic approach with thermal control of thermal management. Thus reducing the costs of repair and maintenance of moving parts and moving parts exposed to corrosive environment, fine dusts both on ground and in space environment. In addition to above such systems are in a unique position to be used both as electrical generation from heat and/or other forms of radiation as well as cooling the structures, thus enhance the applicability of hybrid systems. In our work at AAMU, we, first, established an in house capability to measure Thermal Conductivity, Electrical Conductivity and Seebeck coefficient of nanolayers of any materials on any host substrate. Then we used energetic ions to fabricate nanoscale materials within various host materials. Such system consisting of nanolayers (Quantum Well) of Nanoclusters (Quantum Dots) was then used to synthesis highly efficient thermoelectric generators (TEG). The performance of super lattice thin film thermoelectric device is quantified measuring the dimensionless figure of merit $ZT = S^2 \sigma T/ k$, where S is the Seebeck coefficient, σ the electrical conductivity, k the thermal conductivity , and T the absolute temperature of the super lattice thin film. Our aim is to obtain large ZT values that is to have highly efficient TEG, we have to enhance the electrically conductive, the thermal insulation and increase the Seebeck Coefficient. Some of the materials selected we had to dope the nano-layers by keV implantation of selected species followed by MeV bombardment. In some selected materials systems we formed nano-layered structures by co-deposition followed by MeV bombardment to form nanocrystals. The interaction of QWs as well as the interaction of QDs results in generation of phonon mini-bands reducing the thermal conductivity, while increasing the electrical conductivity resulted in synthesis of TEG with much higher efficiency reported to this date. The materials systems selected for this investigation are Bi_xTe_3/Sb_2Te_3 Nanolayers, $SiO_2/Au_xSiO_{2(1-x)}$ Nanolayers, and multiple periodic layers of $Si_{1-x}Ge_x$ / Si some operate at temperatures

around 300K and some at about 1000K. Following is a schematic example of a TEG superlattice TEG device produced in house.



1- Research sponsored by the Center for Irradiation of Materials, Alabama A&M University and by the AAMURI Center for Advanced Propulsion Materials under the contract number NAG8-1933 from NASA, and by National Science Foundation under Grant No. EPS-0447675.
2- Corresponding author: D. ILA; Tel.: 256-372-5866; Fax: 256-372-5868; Email: ila@cim.aamu.edu
3- Patent filed/Patent Pending

Abstract for IPPW5 Session VII:
Emerging, enabling, and extreme environment technologies; cross-cutting technologies

Real-Time 3D Collaborative Design Simulation in Support of NASA and ESA Planetary Exploration Programmes

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NASA and ESA have a long history of investing in virtual teleoperations, CAD/CAM, aerodynamic visualization and other aspects of 3D modeling and work practice simulation.

Successful recent uses of virtual environments in mission training and operations include the Hubble Telescope repair mission in 1993 and the use of 3D tools such as RSVP and Viz in the current MER missions. To date, however, real-time 3D has been used only infrequently during the concept engineering phase of vehicle and mission design. NASA has instead relied largely on static artist's conceptions and more linear, textual descriptions in documents and teleconference calls. The situation at ESA is largely similar. A problem often observed with the existing modalities is that cognitively complex mission designs cannot easily be communicated and iterated.

DigitalSpace Corporation has been supported by NASA for the past six years to construct an open source, real-time 3D collaborative design engineering and training platform. That platform is now being used to support several projects at NASA and its contractors.

The paper will detail the architecture, successes and limitations of the Digital Spaces (DSS) open source collaborative real-time 3D simulation platform as a rapid prototyping design tool in a number of NASA programs. We believe that the modeling of entire architectures and missions for both in-space vehicles and surface operations is now practical for low to medium fidelity representations of mission trade spaces. Iteration of design concepts before committing greater resources to computer aided design models and physical prototypes may lower costs and provide better designs. Indeed, real-time 3D modeling and simulation may have a role in the full lifetime of a mission being used to ultimately produce tools for day-to-day mission operations.

An example of the planned use of DSS within the context of a prototype Mars exploration platform will be presented.

As this project is open source, we invite participation in the continued development of the platform by NASA, ESA, the academic and aerospace contractor communities.

Keywords: integrated 3D modeling and simulation, trade studies, agent-based work practice simulation, collaborative design simulation, modular training environments

SESSION VIII: Earth Entry, Descent and Landing (EDL) for Sample Return and Crewed Missions (Part 1)

(J. Arnold, B. Foing) Thursday, June 28, 17:15 - 18:35

17:15-17:45	**(Invited) D. Kontinos and M. Stackpoole** *"Overview of Post-Flight Analysis of the Stardust Sample Return Capsule Entry"*
17:45-18:10	**(Invited) B. Foing** "Sample return from the Moon"
18:10-18:25	**P. Desai, and G. Qualls** "Reconstruction of the Stardust Entry"
18:25-18:35	**Discussion**

SESSION VIII: Earth Entry, Descent and Landing (EDL) for Sample Return and Crewed Missions (Part 2)

(J. Arnold, B. Foing) Friday, June 29, 8:30 - 10:00

8:30-8:50	**A. Moisheev, V. Vorontsov, M. Martynov, S. Alexashkin, V. Finchenko** *"Descent Vehicle for Delivery of Soil Samples from Phobos to the Earth"*
8:50-9:10	**(Invited) J. Reuther and E. Venkatapathy** *"Overview of Advanced Technology Development for Orion's Heat Shield"*
9:10-9:30	**(Invited) B. Bryant and J. Corliss** *"Orion Landing System Advanced Development"*
9:30-9:50	**G. Chen, C. De Tong, M. Ivanov, C. Ong, C. Seybold and D. Hash** *"Testing Lunar Return Thermal Protection Systems Using Sub-Scale Flight Test Vehicles"*
9:50-10:00	**Wrap-up discussion: 10 minute on Sample Return Technologies**
10:00-10:15	**Break**

Overview of Post-Flight Analysis of the Stardust Sample Return Capsule Earth Entry

An Invited Paper
Session VIII
International Planetary Probe Workshop # 5
Toulouse France June 25-29, 2007

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Introduction

In the early morning of January 15, 2006, the Stardust Sample Return Capsule (SRC) successfully delivered its precious cargo of cometary ejecta particles to the awaiting recovery team at the Utah Test and Training Range. The SRC returned to Earth at 12.8 km/s, the fastest human-made object to traverse our atmosphere and only the second super-orbital velocity entry since the Apollo Program. In addition to the excitement that comes with the rarity of obtaining extra-terrestrial material, it was a remarkable event in entry physics.

This paper will present an overview of post-flight activities for assessing the performance of the entry system and the analysis tools used to design it. Three sources of information will be leveraged: the recovered SRC thermal protection system, airborne observation of the entry using instruments that provide spectral resolution of the hot SRC and shock layer gasses, and radar signature during the terminal descent stage. The paper will describe the objectives of the post-flight analysis, the information sources, and the methods of analysis. Preliminary results will be shown.

No ITAR restricted information will be contained in this paper, specific content pending NASA approval.

Analysis Objectives

The Orion Thermal Protection System Advanced Development project, Jet Propulsion Laboratory (JPL) Stardust Project and the NASA Engineering and Safety Center (NESC) are jointly performing an integrated reconstruction of the SRC)atmospheric entry to assess the accuracy of the aerodynamic, aerothermodynamic, and Thermal Protection System (TPS) response tools and processes used to design the SRC.

A well instrumented airborne observation of stardust entry during the hypersonic phase provides temporally resolved traces of the entry trajectory, average forebody surface temperature, ablation performance, and shock layer radiative emission.

Since the stardust heatshield was not instrumented, a key component of this reconstruction is destructive post flight evaluation of the heatshield material. Combining the airborne observation data with detailed analysis of the TPS will yield the most complete picture possible of the entry performance. At this stage, 3 cores have been extracted from the heatshield, each core containing information that will aid in this effort.

The objectives of the integrated analysis are to 1) assess aerodynamic performance of the SRC as compared to design prediction, 2) confirm conformance to anticipated trajectory dispersions, 3) assess TPS surface temperature and ablation response as compared to the pre-flight model, 4) assess convective and radiative aerothermodynamic heating models, 5) compare characteristics of flown TPS to arc-jet tested samples to confirm the verification and validation process and 6) provide guidance for Orion flight test objectives and instrumentation approach. Analysis of the recovered hardware is essential to meeting the broader goals stated above. Specific objectives of the hardware analysis are the following:

Forebody Heatshield-

1) Determine unusual surface features indicating off-nominal aerodynamic performance, off-nominal TPS performance, or pre-entry damage.
2) Measure in-depth char and transition layer of TPS at select locations to determine spatially varying integrated heat load.
3) Measure in-depth properties of TPS, e.g. density, chemical composition, and thermal properties, to compare to pre-flight models and arc-jet tested samples.
4) Measure residual bond strength to assess aging effects

Aft-shell-

1) Determine unusual surface features indicating off-nominal aerodynamic performance, off-nominal TPS performance, or pre-entry damage.
2) Measure surface reflectivity as indicator of surface heating, flow pattern, or ablation product deposition.
3) Measure in-depth char and transition layer of TPS at select locations to determine spatially varying integrated heat load.
4) Perform chemical analysis of surface ablation products to assess origin, and chemical analysis of sub layers to identify remaining thermal control coating

SAMPLE RETURN FROM THE MOON

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Introduction: We shall review latest lunar results and the case for future lunar sample returns, as discussed by various ILEWG science and technology task groups:

- New Science opportunity (no global or polar sampling)
- Clues on mantle/lowercrust (South Pole Aitken Basin), polar ice, cometary/meteoritic record
- Technology demonstration preparation for Mars sample return
- Technology demonstrator for lunar ascent vehicle, Earth reentry, and human return vehicle
- A generic lander platform can be adapted to sample return or to a lunar lander /rover fetcher.

Technologies: Technologies can be developed for lunar sample return missions: entry airless bodies, Descent and landing, robotics , Instruments, Sample acquisition, Return and Earth reentry.

They exploit Synergies for various targets (Moon- NEO-Phobos-asteroids-Mars): Planetary entry (differences); Descent/ landing; Rover sample fetcher (similar Moon/Mars); Robotics/sensors; Ascent/return vehicle (local gravity, direct/RdV); Sample capsule/sealing (planetary protection differences); Earth reentry/shield (similar speeds); Sample curation; Sample analysis (isotopic, age, volatiles, organics).

Science questions: Lunar sample returns allow to address fundamental science questions:

- What are the conditions for planetary formation? (bombardment chronology, isotopic dating)
- How does the Solar System work? (Impact basins, accretion, collision)
- Comparative planetology (volcanics, tectonics, cratering, erosion, interior & subsurface)
- What are the conditions for life? (Search for extraterrestrial ice and organics on the Moon)
- Validation of extreme organics and life detection technologies (organics and ice)
- Habitability of Moon (survival and return, life sciences , ecosystems, mini biospheres),
- Search for Early Earth samples

Missions concepts:

Concepts for robotic landers and sample return building on New Frontiers studies, ongoing ESA NEXT studies, and SMART-1 maps of landing sites.

A polar ice sample return would allow new studies, constraining the history of water, volatiles and organics delivery. A Core sample of ice from Permanent shadowed areas would probe the diversity of cometary and water rich asteroids, allowing from retrieved preserved samples to perform isotopic analysis, and search for organics.

A lunar sample return from South Pole Aitken basin material would allow a window into the interior of the Moon and the past, from a huge basin on the far side of the moon created around 4 billion years ago.

New science can come from Sample return from youngest lava basalts in Procellarum, or from Global lunar sampling, or Vertical lithology of craters central peaks.

A sample return demonstrator can also serve for Life sciences and astrobiology, and prepare for ISRU demonstrator.

Early Earth Sample returns might be performed in cooperation with in-situ humans, because of the challenge of drilling and trenching, the elusive signatures for Earth samples, the search for Fossils of organics & ancient life from Early Earth (4 billion years ago).

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RECONSTRUCTION OF THE STARDUST ENTRY

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Introduction: Stardust, the fourth of NASA's Discovery class missions, was launched on February 7, 1999. The spacecraft performed a close flyby of the comet Wild-2. It came within 100 km of the comet nucleus and deployed a sample tray to collect cometary and interstellar particles. Stardust is the first mission to return samples from a comet. Upon Earth return on the morning of January 15, 2006, the entry capsule, containing the comet samples, was released from the main spacecraft and successfully landed by parachute in northwest Utah at the U.S. Military's Utah Test and Training Range (UTTR).

Four hours prior to entry, the Stardust capsule was spun-up and separated from the main spacecraft. The spin-up maintains the entry attitude (nominal 0° angle-of-attack) during coast until atmospheric interface. Throughout the atmospheric entry, the passive capsule relied solely on aerodynamic stability for performing a controlled descent through all aerodynamic flight regimes. At separation, the 46 kg capsule was spun up to 13.5 rpm for entry. A g-switch was triggered after sensing 3 g's, at which point, the drogue timer was initiated. After 15.04 sec, the supersonic drogue chute was deployed (around Mach 1.4), and the main timer was initiated. After 350.6 sec, the main parachute was deployed (at around Mach 0.16). The capsule continued its successful descent until landing. Figure 1 shows the final landing location of the capsule, along with the pre-entry footprint predictions.

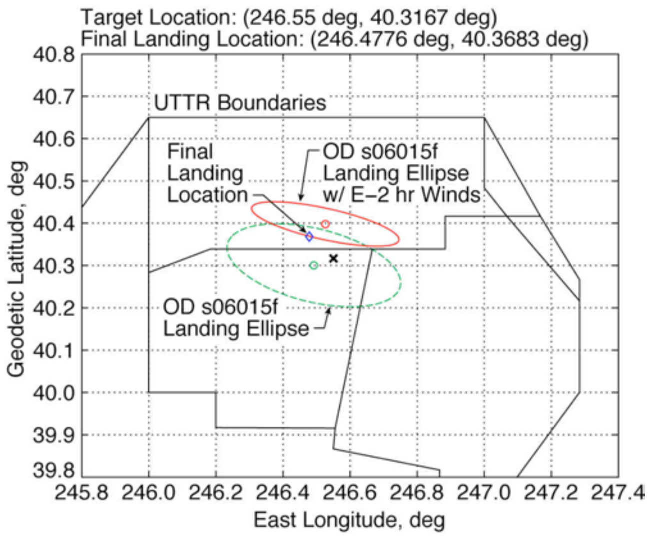

Figure 1. Stardust Capsule Landing Location

This proposed paper will describe the reconstruction analysis performed for the Stardust entry on January 15, 2006 to quantify how well the pre-entry predicted profile compared with what actually transpired. Reconstruction of the entry trajectory parameters will be presented, along with an understanding of the attitude behavior of the capsule during the entry. In addition, a comparison will be made of the reconstructed atmosphere profile experienced during the entry with that assumed prior to entry

DESCENT VEHICLE FOR DELIVERY OF SOIL SAMPLES FROM PHOBOS TO THE EARTH

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ABSTRACT

In accordance with the program of Federal Space Agency of Russia the project of interplanetary mission “Phobos-Grunt” has been developed at Lavochkin Association; the main purpose of the project is delivery of soil samples from Phobos, Mars’s moon, to the Earth. After intake of Phobos soil samples they will be placed into a special capsule inside the vehicle which are to be descended in the Earth atmosphere. The descent vehicle (DV) provides delivery of soil samples on to the Earth surface and protects them from thermal and load action during aerodynamic braking and in the process of the Earth surface encounter as well as from environment action at the place of landing during the period of DV search and evacuation into a relevant research center.

This paper deals with brief descriptions of mission’s stages on delivery of soil samples from Phobos, DV shape and structure, profile of DV descent in the Earth atmosphere, conditions of DV operation at all the stages of interplanetary flight.

The results of analysis and experimental studies are presented in details; they were performed to provide the choice of aerodynamic form and optimal (by the weight) thermal protection of DV surface.

The analysis was made under the following conditions of random enter of DV into the Earth atmosphere at the altitude of 120 km: speed of entry - 11.5…12.1 km/s, angle of entry – minus 33…45 deg.. Aerodynamic and thermal tests were performed at ground experimental facilities: in wind tunnels, at plasma and electric arc plants.

In the result of studies the DV configuration is chosen in the form of a frontal spherical segment and hemispherical bottom part adjoined with the help of a cylindrical spacer. The required DV orientation at aero braking in dense atmospheric layers is provided for presence of rear hemisphere and corresponding position of center of gravity which provide the guaranteed margin of the vehicle static stability.

Dense glass plastic is chosen as a material for frontal and cylindrical surfaces of DV and light material on base of formaldehyde resin with microspheres for protection of DV bottom part.

Manufacturing of thermal protection of frontal and cylindrical surfaces of DV of the material based on organo-silicon polymers is alternatively considered.

Overview of Advanced Technology Development for Orion's Heat Shield

INVITED TALK

IPPW Number 5, Session VIII

June 25-29 Toulouse France

by

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Abstract. The Crew Exploration Vehicle (CEV) Thermal Protection System (TPS) Advanced Development Project (ADP) is a NASA multi-Center activity for providing two heat shield preliminary designs, including the TPS, the carrier structure, the interfaces and the attachments, by CEV PDR. The ADP's primary objective is the development of a single heat shield preliminary design, for the recently named Orion spacecraft, that meets both lunar and LEO return requirements.

Orion will serve as the successor to the current Space Shuttle Orbiter as the primary manned U.S. space vehicle. CEV is planned to begin Low Earth Orbit (LEO) flights to service the International Space Station in 2014 with planned lunar missions to commence in 2018. Its Apollo like capsule will be 5 meters in diameter and enter the atmosphere from lunar return missions at 11 km./sec. The atmospheric entry heating for the Orion heat shield during lunar return missions are expected to be roughly five times more severe than expected during LEO returns. As a result, re-usable TPS solutions, such as those used for the Shuttle Orbiter, are not applicable and ablative material options will be necessary.

While ablative TPS materials have been used for similar entry conditions in the past, most notably during Apollo missions, they have not been the focus of much recent technology development. The effort to create a lunar return capable Orion heat shield is considered a top risk item for the NASA Vision for Space Exploration (VSE) effort due to the low starting TRL (~ 4) of the candidate TPS materials. The TPS ADP was initiated early in the CEV development cycle with the intent to use the testing and analysis of candidate materials in combination with manufacturing demonstrations and heat shield integrated design work to reduce the programmatic risk to Orion and the overall VSE. Currently, the leading TPS material candidate is Phenol Impregnated Carbon Ablator (PICA) - the same material that was used for the Stardust heat shield.

Due to the technical and schedule risks associated a lunar return heat shield, the ADP is pursuing a parallel path design approach, whereby a back-up TPS/heat shield design, that only meets LEO return requirements, is also under development. We are currently considering two candidate TPS materials for LEO only return missions – SLA-561V used for Viking, Pathfinder and MER and now under development for MSL, and Shuttle tiles.

In addition to the obvious maturation of lunar and LEO TPS materials and heat shield systems, we also see the current TPS ADP investments and technology maturation activities to have a major impact on future planetary probe missions. The paper will thus discusses the spin offs from the Orion TPS technology development that are directly applicable to future planetary missions.

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Orion Landing System Advanced Development

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Abstract
The Orion Landing System Advanced Development Program at NASA Langley Research center is designing and testing the prototype landing system for the Orion vehicle. A formal trade study investigated many landing concepts including retro rockets, airbags, deployable crushable structures, and landing legs was conducted. This trade guided the initial activities of the landing system development project. As the Orion requirements and design have matured, so has the proposed landing system. The trade study activities and the evolution of the resulting recommendations for the Orion landing system are described. No ITAR material will be discussed.

Testing Lunar Return Thermal Protection Systems Using Sub-Scale Flight Test Vehicles

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Abstract: A key objective of NASA's Vision for Space Exploration is to revisit the lunar surface. Such an ambitious goal requires the development of a new human-rated spacecraft, the Orion Crew Exploration Vehicle (CEV), to ferry crews to low earth orbit and to the moon. The successful conclusion of both types of missions will require a thermal protection system (TPS) capable of protecting the vehicle and crew from the extreme heat of atmospheric reentry. The development of ablative TPS materials, similar to those used on the Apollo command modules, virtually ceased following the early 1970's in favor of reusable tiles that are currently used on the Space Shuttle. While Shuttle tiles are perfectly suited for the aerothermal environments experienced during return from low Earth orbit, they are not capable of handling the environments Orion heatshield will experience on return from the Moon for which ablative TPS materials are required. Given the lack of development of ablative materials over the past 40 years, the agency finds itself without any efficient, high technology readiness level (TRL) options for the Orion heatshield. As a result, NASA has initiated an advanced development project to raise the TRL of several candidate ablative TPS materials through ground testing, analysis, and an assessment of manufacturing, repair, and operability risks. Although as a part of the development these materials are being tested in arcjet tunnels, the combined lunar return aerothermal environment of high heat flux, shear stress, and surface pressure cannot be duplicated using existing ground test facilities.

To ensure full TPS qualification, a flight test program called Testing Of Reentry Capsule Heatshield, or TORCH, has been proposed to specifically test the new TPS materials and heatshield construction techniques under the most stressing combination of lunar return aerothermal environments by using sub-scale Orion capsules. Each TORCH capsule is a 2-meter diameter, geometrically scaled model of the Orion crew module with a fully functional hypersonic guidance and control system. Each capsule also has a highly instrumented aeroshell, traceable in design to the full-scale Orion capsule. TORCH flight tests are designed to launch on expendable launch vehicles onto sub-orbital trajectories and land at the Woomera Test Range in Australia so that the TPS can be recovered for analysis.

The driving design requirements of the sub-scale TORCH vehicle are presented as well as a description of the TPS instrumentation suite slated to conduct in-flight measurements. The design highlights of the TORCH spacecraft are described along with a discussion of the design trades that led to a cost effective flight system design versatile enough to test the edges of the TPS flight envelope as well as other Orion subsystems. Also included is an overview of the trajectory design methodologies used to ensure that a sub-scale vehicle can achieve the same combined aerothermal conditions as the full scale Orion capsule. Finally, launch vehicle options for the various flight test missions are discussed.

Further details on the TORCH flight system and trajectory design can be found at the poster session.

SESSION IX : Future outlook
(J-P Lebreton, J. Cutts)
Friday, June 29, 10:15-12:30

10:15-10:35	**A. Coustenis, T. Spilker, K. Baines, B. Bienstock, P. Plotard** *“Highlights of new mission concept development”*
10:35-10:55	**P. Beauchamp, E. Kolawa, N.Cheatwood, A. Ball, L. Peltz, Th. Blancquaert** *“Highlights of technology developments”*
10:55-11:15	**S.Hubbard** (chair), Participants to be confirmed *“Prospects for international collaboration”*
11:15-11:30	*Overall discussion*
11:30-12:00	*Poster awards*
12:00-12:15	*Next Workshop, IPPW-6*
12:15-12:45	*Closing remarks*
13:00	*Workshop adjourn*

Poster session II : Mission concept studies, and science drivers of technology, and sample return – Venus and Mars
(B. Bienstock, K. Baines, P. Plotard)

Poster 2.1	**M. Forrest, et. al.,** *"Mars Anti-Polar Lander Network Mission: EDL and Science Station Design"*
Poster 2.2	**S. Otter, et. al.** *"Spectroscopy at the Martian Surface - Modelling and Future Instrumentation"*

Mars Anti-Polar Lander Network Mission: EDL and Science Station Design

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The current paper presents work awarded by the California Space Grant Foundation that encourages collaboration between the San Jose State University Aerospace Engineering Program and NASA Ames Research Center's scientists to design a Mars polar entry probe and surface science station. The main objective of the Spartan Atromos mission is to demonstrate the technological and economical feasibility of micro-class probes for network missions. With the mass constraint of 20 kg, Spartan Atromos will demonstrate its robustness by operating on the unexplored Martian Polar Regions.

For the analysis and design of the entry probe and science station, an N^2 diagram was used to decouple the system into individual subsystems and to identify key variables that affected the overall system design. For the Entry, Descent, and Landing (EDL) phase of the mission, a trajectory was computed that takes into account the entry phase of the mission up to parachute deployment. By selecting key points of interest along the entry trajectory, high-fidelity computational fluid dynamics (CFD) analysis was performed on the unique entry probe configuration to characterize the aerothermal environment for thermal protection system (TPS) sizing. The commercial CFD software that was used, ESI-FASTRAN, uses a 3D Navier-Stokes flow solver with finite rate chemistry and thermal non-equilibrium. For the descent phase of the mission a supersonic parachute design was explored. This design incorporated canopy configurations similar to that of Disk-Gap-Band (DGB) and ring slot configurations. The deployment system was designed for a deployment Mach number of 2 to 3 and a terminal descent velocity at the surface of 20 m/s. This canopy configuration was constructed and tested for low speed stability and exhibited satisfactory performance. For the landing phase of the mission, the Science Station will be severed from the parachute and allowed to free-fall to the surface. A deployable rigid landing structure was developed to decrease deployment complexity and increase functionality for the science station payload and solar panels. Analysis was performed on Pro-Engineer/Mechanica to determine the structural integrity of the landing spokes. A drop-test was performed to benchmark the numerical analysis and prove the design concept for a new type of landing system.

Once landed, the science station is designed to perform *in-situ* measurements on the Martian surface. The design of the science station was broken down into four subsystems: thermal, structure, power, and payload. By using the N^2 diagram to analyze the system; two distinct candidate design concepts were developed and studied to meet the mission constraints and objectives. The concept of vertically-oriented solar panels required due to the low incidence angle of the sun in the Northern polar regions was studied. A Deployable Mars Tower (DMT) that serves as a sensor array to measure temperature, pressure and velocity gradients in the boundary layer near the surface was also designed and built. DMT uses elastic memory composites which, upon heating, return to their original fabricated shapes and allow for fairly complex structures to be deployed with minimal active systems.

SPECTROSCOPY AT THE MARTIAN SURFACE – MODELLING AND FUTURE INSTRUMENTATION

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As part of ESA's Aurora programme the ExoMars mission plans to send a descent module containing a rover and a long term base station to the Martian surface. One of the instruments proposed for the science payload is a miniature spectrometer, currently in development, with a proposed functional wavelength regime of ~ 200 – 650 nm (UV to visible). This instrument will be used to quantify *in situ* for the first time the UV spectrum at the Martian surface. It is hoped that through analysis of these spectra and iterative fitting of a radiative transfer model, it will be possible to constrain accurately the optical properties of the dust suspended in the Martian atmosphere at UV wavelengths. Computer modelling of the radiative transfer process through the Martian atmosphere and the creation of a spectral database of proposed Martian surface-analogue samples will be incorporated into the testing and calibration of the spectrometer. Study and characterisation of the Martian atmosphere and its attenuation of this region of the electromagnetic spectrum also has important astrobiological implications, such as quantifying the levels of UV insolation at the Martian surface with regard to life surviving under such conditions. With human exploration of Mars anticipated for the future, the ability to measure and predict the levels of potentially hazardous radiation a manned mission would encounter is imperative.

Poster session III : Entry, Descent and Landing Concepts for Current and Future Missions beyond Earth
(M. Wright, A. Ball, W. Lee)

Poster 3.1	**G. Gelly** *“Guidance Viewpoints of Mars exploratory missions”*
Poster 3.2	**R. Lorenz** *“Identification of Atmospheric Turbulence Parameters in Dynamics Measurements on Probes and Balloons”*
Poster 3.3	**M. I. Gritsevich** *“Luminosity of gas at the entry of probes and natural cosmic bodies into planetary atmospheres”*
Poster 3.4	**A. Howard, A. Colaprete** *“Miniturization of atmospheric entry probes options for future planetary exploration missions”*
Poster 3.5	**C. Tang, M. Wright, K. Edquist, and A. Dyakonov** *“Simulation of Aftshell Features on the Mars Science Laboratory Entry Capsule”*
Poster 3.6	**R. H. Ramos** *“Status of ExoMars Mission Analysis and Design: Entry, Descent and Landing”*

POSTER SESSION: GUIDANCE VIEWPOINTS OF MARS EXPLORATORY MISSIONS

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Although efforts are underway to continue improvements in reliability and sensitivity of robotic planetary probes, they will not, in the foreseeable future, be able to match the examination and analysis capabilities available on Earth. One solution to this dilemma is to retrieve planetary samples for analysis on Earth. Within the frame of interplanetary missions, as considered by ESA/ESTEC in the AURORA exploratory program, the recovery of either a Mars soil sample collected by a robotic mission, or a crew in the frame of a manned mission, will require to master both the descent to Mars ground and the following ascent phase. During the past few years, Astrium Space Transportation has demonstrated its technical capabilities on Entry, Descent and Landing Systems as well as on Planetary Ascent Vehicles.

Entry, Descent and Landing Systems: different entry vehicles such as ARD or Phoenix have been developed by Astrium Space Transportation and have successfully landed on Earth and several Entry and Descent Systems (such as Huygens) have been developed to enter the atmosphere of other planets. The next logical step is then to perform a Soft-Landing on another planet. In this ambitious perspective, Astrium Space Transportation has realized system studies on a vehicle able to successfully perform a soft-landing on Mars: various soft-landing strategies (sky-crane manoeuvre, touchdown with vented airbags…) as well as several propulsive architectures (liquid, solid, throttleable, gimballed…) have been considered for this purpose. For such missions, one can find a wide range of possible ways to solve the guidance problem whose main objective is to perform a pin-point landing on a safe landing site. The relative simplicity of the dynamic equations of a lander (atmospheric effects are negligible at low velocity) allows an explicit resolution of the guidance two-point boundary problem. Then, most of the explicit guidance algorithms studied and developed by Astrium Space Transportation along the years could be applied.

The well-known Gravity Turn approach for performing a landing is based on a quite simple concept that leads to a very poor position accuracy at landing and thus does not allow hazard avoidance retargeting. Thus, one has to turn to simple explicit optimization methods such as the Apollo E guidance, the Bilinear Tangent law, the Chandler scheme, etc. These explicit schemes aim at autonomously reaching the target by self-building a steering law that is updated, on board, according to the current flight conditions. They can also be used for trajectory planning in case a re-targeting is commanded during the descent. More sophisticated methods can be considered such as the Optimal Command steering law, a predictor corrector, collocation methods or even more exotic methods such as Neural Networks trained to solve this specific trajectory problem. The main added value in this case is the potential gain in performance (propellant saving) and robustness towards perturbations and landing target changes. The performances of all these algorithms have been assessed with a 6 dof simulator developed with Simulink.

Mars Ascent Vehicle: simultaneously, a dedicated ESA/ESTEC study (ITT 4333) was led by Astrium Space Transportation as prime contractor, focusing on the GNC for a Mars Ascent Vehicle. In this frame, guidance algorithms that allow the ascent vehicle to successfully retrieve the samples collected from the Martian ground have been investigated.

Traditional ascent guidance algorithms have been relatively simple. The common methodology is to operate in ‘open-loop’ mode during the (early) high dynamic pressure portion of flight and then, based on a pre-determined time or event, switch to a ‘closed-loop’ vacuum guidance scheme which operates on the premise that aerodynamic forces can be neglected. The ‘open-loop’ mode typically makes use of pre-loaded tables of Euler attitude commands versus time or speed. The ‘closed-loop’ logic is based on explicit formulas and simplified dynamics that result in a semi-analytical solution for the optimal steering angles. Then, the performances of three guidance schemes that have been developed in the frame of the AURORA program were studied. The first one is an implicit scheme base on look-up tables (the interpolation parameter being the relative velocity), the next one is a predictor-corrector and the last one is an explicit guidance inspired by the bilinear tangent law.

IDENTIFICATION OF ATMOSPHERIC TURBULENCE PARAMETERS IN DYNAMIC MEASUREMENTS ON PROBES AND BALLOONS.

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Introduction: Dynamical measurements (accelerometers, Doppler tracking, and more elaborate data such as gyros, optical data and so on) are always acquired to some extent on planetary probes to document their descent. However, the interpretation of such data into properties of the atmosphere has not always been carried through to the fullest extent possible.

The challenge is to separate the signatures of atmospheric motions (shear, eddies etc.) from the self-excited motions of the vehicle itself. Spectral analysis of dynamic motions is one approach that has been applied e.g. to Galileo data, although the separation of swinging and spinning motions is not always clear-cut.

Some insight into the separation of externally-excited motions has been recently obtained with terrestrial balloons, which indicate particular excitation of the pendulum mode in cloud turbulence. In the dynamic sense, a probe or balloon is much like a seismometer. It is a damped resonant system and typically exhibits a flat low frequency response, with a roll-off towards high frequencies., but in any case generally 'red' in spectral colour. In contrast, when being specifically excited by external air motions, a spectral peak emerges at around the natural frequency.

In addition to spectral analysis, the statistical moments of the motion provide some insight. In particular, the kurtosis of the distribution (the normalized third moment) is a useful measure of non-Gaussianity, and indeed is used in turbomachinery monitoring for this purpose, to identify incipient failure. The Huygens data show a pronounced excursion in kurtosis at the same time as the spectral analysis indicates turbulence – interestingly the kurtosis event is not accompanied by a corresponding increase in amplitude.

We will discuss the Huygens results, from other data as well as the SSP Tilt sensors, and explore the application of these analysis techniques to other probe or balloon missions. It is obvious that the interpretability of these data is facilited (if not enabled) by model tests – wind tunnel and especially free-flight trials. Such testing should be factored into the development of future probe/balloon mission, for scientific reasons even if the technical demonstration of the system does not require it.

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LUMINOSITY OF GAS AT THE ENTRY OF PROBES AND NATURAL COSMIC BODIES INTO PLANETARY ATMOSPHERES

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Previously, it was considered in the literature that luminosity of a body at its movement in the atmosphere with large speed is defined only by radiation of vapor of a body material, arising owing to evaporation of its surface. Therefore change of body mass on a shone sector of its trajectory was defined by integration of luminosity. This photometric formula long time was used as a unique way to determine of extra-atmospheric mass. On the other hand, the mass of a meteoric body characterizes height and intensity of meteor braking in the atmosphere. The essential divergence of mass values, received by these two ways, was marked in a number of works on an example of fireballs from the European network and from the Prairie network, USA. Almost always the photometric mass on the order and more exceeded the mass defined on intensity of braking. In the report, other treatment of luminosity of high-speed object in the atmosphere, not connected with its evaporation, is resulted.

At a flow past body in a rate of the continuous medium, the basic contribution to luminescence gives radiation of heat atmospheric gas near the body. Intensity of luminosity of an object in the optical range can be received by direct calculation of the flow about the body taking into account the heat transfer by radiation.

In the report, results of such calculation with the assumption of equilibrium flow in a shock layer and constant temperature of a body surface are presented. A simple model of a hypersonic flow past body is used. The form of the body is spherical, and the total flow is calculated as a stream past a circular cylinder with spherical nose. Radiation of the stream is calculated in the optical thin approach.

Results for luminosity of moving object are presented in the form of power dependence on the size of a body, its speed and density of the atmosphere (or height of flight). In the report, ranges of applicability of this formula on the specified variables are resulted.

Comparison of calculations with observational data for the Lost City fireball, received from the Prairie network, USA, in 1971 is resulted. Satisfactory conformity of calculations and observations, mainly for the bottom part of the trajectory is shown. Distinction of data in the top part of a trajectory arises owing to the assumption of equilibrium current in a shock layer past the body and in a trace behind it.

MINITURIZATION OF ATMOSPHERIC ENTRY PROBES: OPTIONS FOR FUTURE PLANETARY EXPLORATION MISSIONS

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Introduction: Atmospheric and planetary science has traditionally been performed by large (100kg-1000kg) probes and probe/lander combinations. This poster explores the small (0.5kg-10kg) probe concept and discusses the potential role that small, focused science platforms can play in the current vision of solar system exploration.

Reducing the size, complexity, and mass of entry probes has the potential to greatly reduce the cost of collecting atmospheric/planetary science data. Additionally, small probes have the potential to provide a scientifically focused platform that can be produced and deployed in relatively large quantities, resulting in an increase in spatial and temporal resolution of the data delivered by probe missions. However, there exist certain technological barriers that need to be addressed before such missions gain wide spread acceptance in the solar system exploration community. This poster presents a summary of the small probe concept, as well as highlighting some of the issues and advantages of such a platform.

Poster: **S**pecifically, the poster will (1) summarize the motivation for atmospheric entry probes and discuss the potential role that small (0.5-10 kg) probes can play in space exploration, (2) list advantages and disadvantages of the microprobe concept, (3) list potential destinations and cogent science objectives suited for microprobes, (4) list some technological challenges which need to be addressed before small probes can become viable and (5) present several conceptual mission designs.

SIMULATION OF AFTSHELL FEATURES ON THE MARS SCIENCE LABORATORY ENTRY CAPSULE

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Introduction: The Mars Science Laboratory (MSL) entry capsule will land a large rover (weight greater than 800 kg) on the surface of Mars in 2010. The purpose of this mission is to place a rover at a site of scientific interest between +/- 60 degrees latitude with a footprint smaller than 10 km. To achieve the desired landing accuracy and altitude requirements, the probe will use a reaction control system (RCS) to adjust the flight path for uncertainties in the predicted entry states and variations in atmospheric conditions. Due to the large aeroshell and the demanding entry trajectory, it is expected that the flow will transition to turbulent as the probe descends through the Martian atmosphere. Consequently, the vehicle will experience surface heating and shear stresses that are significantly higher than previous Mars missions (Viking 1 and 2, Mars Pathfinder, and Mars Exploration Rovers Spirit and Opportunity). The use of the aftshell RCS thrusters may also result in high heating on certain areas of the backshell. Hypersonic Navier-Stokes simulations using *LAURA* and *DPLR* have predicted surface heat flux, which includes uncertainties, as high as 76 W/cm^2 on the backshell and 38 W/cm^2 on the parachute cover for a smooth outer mold line (OML) model of the MSL configuration. Since the actual vehicle will have various protrusions and cavities on the aftshell (for example, antennas, covers for the RCS, …), the heating around these geometric features may be even higher than the predicted smooth OML results. The goal of this paper is to include these backshell features in the simulations and to study their effects on the aerothermal environment. The solutions from these calculations will be used to design an aftshell thermal protection system (TPS) capable of surviving the harsh environment of the proposed mission.

Status of ExoMars Mission Analysis and Design – Entry, Descent and Landing

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ExoMars is ESA's current mission to planet Mars. The probe is aimed for launch between 2011 and 2013. The project is currently undergoing Phase B studies under ESA management and Alcatel Alenia Space project leadership. In that context, Deimos Space is responsible for the Mission Analysis and Design for the interplanetary and the Entry, Descent and Landing (EDL) activities. Within this contract, Analyticon Ltd. is responsible for the sizing and analysis of the Descent and Landing system.

The mission analysis and design of the EDL comprises the flight from the separation of the Descent Module (DM) from the carrier up to the landing onto the Mars surface. The current mission baseline is based on a Soyuz-Fregat launch from Kourou in 2013 of a spacecraft composite bearing a Carrier and DM. A back-up option is proposed in 2015. Additional scenarios with an orbiter have been considered, covering dual launch with Soyuz and single launch with Ariane 5 for launch opportunities at 2013 and 2015.

This paper regards the flight mechanics of the EDL phases, with particular attention to the characterization of the mission requirements and constraints that drive the mission feasibility.

At this stage of project, the design of the EDL phases is driven by the flexibility in terms of landing site and the very stringent requirement in terms of landing accuracy. Current specification states that the DM must be able to land during daylight within the latitude band (15ºS,45ºN) at a maximum altitude above the MOLA areoid of 0 m with a demanding landing accuracy. Two options for the landing systems are analyzed: vented and non-vented airbags.

The envisaged EDL concept will be based on the following general sequence: after separation, the DM enters the atmosphere and deploys a two stage parachute system. The heatshield is released and, in case of non-vented airbag, the Lander is lowered. Vertical solid or liquid retrorockets are ignited to perform the final breaking. The airbags (vented or non-vented) are deployed and the Lander free falls after release

Innovative approaches in terms of worst-case philosophy have been designed in order to cope with that variability and to provide a global picture of the mission capabilities in terms of landing site reachability. The concept of the entry corridors has been extended to a planetary level (Global Entry Corridor) for identification of realistic worst cases with direct mapping of landing site location with mission performance.

These entry corridors are combined with the filtering of the landing sites based on terrain characteristics, such as altitude or maximum terrain slope in order to provide a global view of the reachability of each landing region. It will be matched with the scientific regions of interest once the Exomars scientific community defines them.

The design of the Descent and Landing phases is driven by the mass and size of the Lander (< 480 kg), the limited mass budget for the overall DM due to launcher capability, the safe separation between the DM and the frontshield and the mass margins philosophy. The D&L system mass and parameters must be traded against the height-loss performance to ensure that terminal descent can be reached in time to deploy the airbags and land.

Parameterisation of the DLS sizing is required to properly support the System activities. It is particularly challenging for the DLS components as a result of the strong non-linear interactions between the masses. To address this problem, an efficient parametric analysis tool has been developed which converges on a robust self-consistent solution for each of a large grid of possible input parameter values. This tool has been widely used to assess the balance between the descent under parachutes, powered descent and landing with airbags phases.

The design of the Entry, Descent and Landing phases and the analysis of the different options for the EDL systems (single or two stages parachutes, fixed or modulated thrust, vented or non-vented airbag…) are used to support the specification of the EDLS (parachute, airbag, retro-rockets, frontshield).

Poster session IV : Technology Systems, Electronics, Instruments and Sensors, Communications and Batteries
(P. Beauchamp, Th. Blancquaert)

Poster 4.1	**F. De Filippis, A. Del Vecchio, A. Martucci; R. Savino** *“Flat faced water calorimetric probe for heat flux measurements in plasma wind tunnels”*
Poster 4.2	**A. Morris** *“In-situ monitoring of compounds in planetary atmospheres using a minaturised field assymetric ion mobility spectrometer (FAIMS)”*
Poster 4.3	**G.W. Hunter, R.S. Okojie, M. Krasowski, G.M. Beheim, G. Fralick, J. Wrbanek, P. Greenberg, P.G Neudeck, and J. Xu** *“Microsystems, Space Qualified Electronics, and Mobile Sensor Platforms for Harsh Environment Applications and Planetary Exploration”*
Poster 4.4	**R. Trautner** *“The ESA Planetary Science Archive - A Source of Science and Engineering Data from European Planetary Missions including the Huygens Data Archive”*
Poster 4.5	**J. Schlee** *“Thermal Protection System (TPS) Embedded Sensor Technologies”*
Poster 4.6	**Y. Lin** *“To preserve and to protect - planetary protection considerations for planetary probes and in situ instruments”*
Poster 4.7	**University of Idaho Team ThermaSense** *“Wireless sensors in thermal protection systems”*

FLAT FACED WATER CALORIMETRIC PROBE FOR HEAT FLUX MEASUREMENTS IN PLASMA WIND TUNNELS

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Abstract: A research activity has been carried out at CIRA, in collaboration with the University of Naples, to develop a new, water cooled, calorimetric probe, able to evaluate the stagnation point heat flux by means of the measurements of the mass flow rate and the temperature difference between the inlet and the outlet of the calorimeter's water circuit. The geometry of this probe is a rounded flat cylinder of 10 cm diameter. The structure is composed by an external copper shell and a core calorimetric sensor that is exposed to the heat flux at the centreline of the calorimeter surface (stagnation point). The cooling systems of the probe and of the calorimeter are being designed to guarantee suitable water temperatures according to the structural properties of the selected materials. The development and realization of this device has been carried out, and specific tests have been done with the plasma facility available at the University of Naples and in 70 MW hypersonic facility PWT SCIROCCO at CIRA. The objective of these tests was the verification of the thermal-structural resistance of the system in real hypersonic environment and the calibration of the realized instrument (at different flow conditions). Comparisons have been carried out with other heat flux techniques, such as Gardon gauges, and with the numerical simulations that have been done. The paper deals with the aero-thermal design of the probe, with the solutions of the various problems, and with the experimental and numerical results.

IN-SITU MONITORING OF COMPOUNDS IN PLANETARY ATMOSPHERES USING A MINATURISED FIELD ASSYMETRIC ION MOBILITY SPECTROMETER (FAIMS)

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Owlstone Ltd. [1], a spin-out company from the University of Cambridge, has overcome the limitations associated with traditional chemical sensors with the development of a ground-breaking solid-state sensor whose operational parameters can be fine-tuned to detect a wide range of volatile chemicals in extremely small quantities. The Field Asymmetric Ion Mobility Spectrometer (FAIMS) outperforms conventional Ion Mobility Spectrometer detectors in the critical areas of size and weight, reliability, sensitivity, response speed, power consumption, versatility and cost of manufacture.

Owlstone Ltd. have linked up with the Planetary and Space Sciences Research Institute at the Open University to to evaluate the analytical suitability of the sensor and the specific design of the sampling interface for its possible use for planetary science. Owlstone Ltd. have funded a matched PhD studentship for Andrew Morris, which begun in October 2007. The development of a fast Gas Chromatography (GC) interface, further enhancing the selectivity of the sensor, will be a primary focus of the project.

The theory behind the technology and proposed future developments will be discussed along with its potential application for planetary exploration.

References:

[1] www.owlstone.co.uk

Microsystems, Space Qualified Electronics, and Mobile Sensor Platforms for Harsh Environment Applications and Planetary Exploration

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NASA Glenn Research Center (GRC) is presently developing and applying a range of sensor and electronic technologies that, while initially developed for other applications, can enable future planetary missions. These include high temperature sensors for Venus missions, space qualified instruments and electronics, mobile sensor platforms, and microsensors for detection of a range of chemical species. The fundamental approach is to develop smart systems for a range of harsh environment and aerospace applications.

World-leading development in harsh environment sensors and electronics is on-going and uniquely positioned to contribute to future Venus missions. While the high temperature electronics capability is covered in another paper at this conference, this paper includes a discussion on a wide range of sensor technology for in-situ Venus measurement applications. This includes pressure sensors, an anemometer, and a resistance temperature differential sensor integrated on a single chip to make it a weather sensor chip. The chemical environment can be measured using a MEMS based chemical sensor array (High Temperature Electronic Nose) previously demonstrated for engine emission sensing applications. Physical measurements on surfaces can be obtained by multifunctional thin film sensors providing surface temperature, strain, and heat flux in a single MEMS based sensor. Overall, these base technologies, being developed for engine applications, can have a significant effect on possible Venus missions.

NASA GRC also has extensive experience in space qualified electronics integrated with instruments systems and mobile platforms. This technology base can be applied to a range of space applications. For example, the Material Adherence Experiment (MAE) flew on the Mars Pathfinder Sojourner in 1996 and measured the effects of dust on the rover's solar panel. This group has also produced the Materials International Space Station Experiment 5 (MISSE-5) Forward Technology Solar Cell Experiment. MISSE-5 is currently operating successfully while mounted external to the International Space Station. MISSE-5 can support up to 36 sensors and has been successfully sending data to Earth with neither data corruption or interruption despite high radiation fluence. This electronics capability is being combined with mobile sensor platforms for sensor placement. This work also included methods for communicating between roving platforms and a central command location. This work leverages commercially available equipment to miniaturize existing sensor platforms and produce mobile platforms that are planetary exploration compatible. The major thrust of this work is to produce systems of sturdy, simple design meeting a technology gap in methods to move sensors from one location to another. These mobile sensor platforms can be integrated with a range of instrumentation as well potential alternate modes of locomotion for potential planetary exploration.

A range of microsystems based chemical sensors have also been developed for applications such as fire detection, leak detection, EVA, and environmental monitoring. These microsensors have been demonstrated on a range of aerospace applications. This technology utilizes basic microsensor platforms that can be modified as needed for a given application. A range of gases of general planetary exploration interest can be measured including methane, ammonia, carbon dioxide, oxygen, and hydrogen. This development also includes a "Lick and Stick" sensor package featuring sensors, power, signal conditioning, and telemetry on a near postage stamp size unit. Also being developed is a microfabricated particle classifier. The combination of a range of chemical species measurements with particle classification is a significant potential tool to broadly characterize a planetary surface.

THE ESA PLANETARY SCIENCE ARCHIVE – A SOURCE OF SCIENCE AND ENGINEERING DATA FROM EUROPEAN PLANETARY MISSIONS INCLUDING THE HUYGENS DATA ARCHIVE

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Introduction: Scientific and engineering data from ESA's planetary missions are made accessible to the world-wide scientific community via the Planetary Science Archive (PSA), see [1]. The PSA offers several online services incorporating search, preview, download, notification and delivery basket functionality. The PSA data repositories contain science and selected housekeeping data from various ESA missions, including data from Europe's first planetary entry probe, Huygens. The Huygens archive is a coordinated effort from all Huygens instrument teams, the PDS atmospheres Node [2] and the PSA. Other datasets that are highly relevant for planetary entry probe science and engineering are those based on atmospheric investigations performed at Mars and Venus. Data preparation for ongoing missions are underway, and data from new planetary missions such as Exomars will be added as they become available.

Data Services: Primary emphasis of the archiving efforts are on long-term data and knowledge preservation. Scientific users of the data can access the data online using several interfaces:

Classical Interface
Map-based Interface
Dataset Browser Interface

The Classical Interface allows parameter based queries, the Map-based Interface the specification of a region-of-interest and the visualization of query results and the Dataset Browser Interface facilitates the direct browsing and access of the dataset content. Each dataset contains documentation and calibration information in addition to the scientific or engineering data.

Data Processing: All science data are prepared by the corresponding instrument teams. Selected housekeeping data is prepared at ESA/RSSD. PSA staff supports the instrument teams in the full archiving process, starting from the definition of the data products, definition of data labels towards the validation and ingestion of the products into the archive. To insure a common archiving approach for all ESA's planetary missions as well as to providea same data quality and standard for end users, a dataset validation tool was developed for supporting the instrument teams in the validation of the syntax of their datasets before delivering to the PSA. In a next step, a further validation is envisaged to ensure correctness, completeness and cross correlation of all information, label and data content, within a dataset.

Archive Approach: All data are compatible to the Planetary Data System (PDS) Standard and the PSA staff work in close collaboration with the PDS Staff. New areas of data exploitation as e.g. interoperability are being explored. Also major contributions are done toward the internationalization of planetary data standards, see [3]. A PSA advisory body has been set up that meets regularly to review the progress on defined requirements.

References:
[1] PSA Home Page, http://www.rssd.esa.int/psa
[2] PDS Atmospheres Node, http://pds-atmospheres.nmsu.edu
[3] IPDA Home Page, http://planetarydata.org

Thermal Protection System (TPS) Embedded Sensor Technologies: Poster Abstract

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Abstract

Planetary probes are used to explore the solar system, search for clues to the origin of life, and study phenomena not easily observable from Earth. The Thermal Protection System (TPS) protects probes from the harsh atmospheric entry environment. To date little engineering instrumentation has been embedded into Thermal Protection Systems due to complexity and risk concerns. The Galileo probe contained ablation sensors embedded into the TPS. These ablation sensors revealed inconsistencies in the engineering recession models. Future missions, such as those under consideration to Saturn, Venus, and Mars, will benefit from further development of TPS embedded sensors. The goal of this research is to examine the variety of measurements and identify which sensors to embed or place in the proximity of the TPS. These measurements will provide valuable data that will not only help characterize the performance of the Thermal Protection System, but will also contribute to a better understanding of the probe entry environment.

To preserve and to protect: planetary protection considerations for planetary probes and in situ instruments

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Interplanetary probes have traditionally been used to conduct detailed studies on other planets within our Solar System and to look for life. So far, the search for life has been focused on Mars. Future interplanetary missions will allow us to explore other extraterrestrial bodies with the help of emerging new technologies to design better spacecrafts and instruments that can operate in extreme environments. While it is important to design and build better probes and instruments to conduct more sophisticated scientific investigations, we must also ensure these missions meet a set of strict biological contamination requirements, generically referred to as "Planetary Protection Requirement". Planetary protection policies have been established to preserve planetary conditions to ensure that the scientific investigations and the interpretation of scientific measurements are not confounded by Earth life that might have been transferred to the planet of interest. They are also in place to protect Earth and its biosphere from potential extraterrestrial sources of contamination. Depending on the type of missions, there are specific planetary protection requirements. To meet those requirements, certain bioburden reduction processes will be used. These processes could affect the spacecraft and instrument functions during operations. Therefore, planetary protection strategies should be considered at the early stage of a mission planning that includes mission architecture plan, spacecraft and instrument design, and material selection. This paper will address some of the issues and potential challenges involved in preparing modern spacecraft and instruments for such missions.

WIRELESS SENSORS IN THERMAL PROTECTION SYSTEMS

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Abstract: The goal of the TPS Wireless Sensor project is to design, build, and test a wireless system that can be integrated into TPS materials on planetary entry probe missions. Currently if sensors are embedded within Thermal Protection Systems, the sensors must be fully wired to carry power, commands, and data from the sensors to a data collection system within the spacecraft. This current system adds unnecessary weight and has the potential to increase the flight risk. As a result sensors have not been frequently used. To save mass, complexity, and reduce risk, a fully wireless system has been designed and tested.

Several different wireless transmission methods were researched including radio frequency, infrared, and near field electromagnetic coupling. Preliminary results show that an RF transmitter was a reasonable choice, because there would not necessarily be line of sight for a IR transmitter and electromagnetic coupling is an underdeveloped technology. The RF protocol selected was ZigBee, a protocol used mostly by industry to transmit sensor network data for low power applications. To demonstrate proof of concept, a wireless TPS temperature measurement system comprising of four thermocouples was designed, built, and tested at the NASA Ames X-jet facility. Tests were conducted using different RF transmitter antenna configurations, power settings, heat fluxes, and transmission distances. Test results showed that the wireless system could transmit out of the environment without much interference. There was some packet loss and other problems involved with the thermocouple bonding in the TPS but the transmission link was found to be strong.

Poster session V : Mission Concept Studies and Science Drivers of Technology- Giant Planets and Titan
(A. Coustenis, T. Spilker)

Poster 5.1	**G. Bampasidis, X. Moussas** *“Future probe instrumentation in Titan : Seismograph?”*
Poster 5.2	**G. Bampasidis, A. Coustenis, X. Moussas** *“Titan : Determination of the local tectonic field at the Titan lake observed from the Cassini flyby on february 2007”*
Poster 5.3	**K. Sugiyama, M. Odaka, K. Nakajima, Y.-Y. Hayashi** *“Numerical Modeling of Moist Convection in Jupiter’s atmosphere and future Jupiter probe mission”*

FUTURE PROBE INSTRUMENTATION IN TITAN: SEISMOGRAPH?

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Abstract. The only extraterrestrial body we have seismic data for is the Earth's Moon. In this work we examine the advantages and disadvantages of the placement of a seismograph on Titan's surface, as part of a landing probe payload. Thus, we study the heritage of Earth-Moon experiments, technical requirments and restrictions (like weight, power, data transmission), possible areas of location and major scientific goals of a seismic experiment and its use in future missions.

Keywords: Titan, Moon seismic experiment

TITAN: DETERMINATION OF THE LOCAL TECTONIC FIELD AT THE TITAN LAKE OBSERVED FROM THE CASSINI FLYBY ON FEBRUARY 22, 2007

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Abstract. Images from the Huygens' descending phase revealed different and strange features on Titan's surface such as craters, dunes, volcanoes, ridges, lakes, mountains, etc. Moreover, the Cassini orbiter will execute 45 flybys of Titan, so more details are certain to appear. The Cassini's radar instrument during a near-polar flyby on February 22, 2007, shows a big island or a peninsula in the middle of one of the larger hydrocarbon lakes imaged on Titan. We can consider the interaction between the local tectonic field and atmospheric phenomena which can shape this island by studying its drainage.

The main attribute of this work is to quantify the hierarchy of stream segments according to the ordering classification system proposed by Horton & Strahler in the observed drainage. In this system, the channel segment which begins from the head of the stream is assigned by the value 1 and called first-order stream. When two first-order streams come together, they form a second-order stream, two second order streams formed a third order stream, and so on. After having measured the length of the stream segments and its basins, it is possible to estimate the liquid budgets of this drainage and the hydrocarbon lake sypply.

Analysis of these data reveals some interesting assumptions about vertical tectonic movements of the area. In addition, the measurement of drainage density provides a useful numerical measure of landscape dissection and runoff potential.

We will present these considerations in the framework of future space missions to Titan with an orbiter. Higher resolution images from a laser or radar altimeter on board, will define better the estimated measurments.

Keywords: Titan, Huygens-Cassini mission, Stream classification system

NUMERICAL MODELING OF MOIST CONVECTION IN JUPITER'S ATMOSPHERE AND FUTURE JUPITER PROBE MISSION

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Introduction: The averaged structure of Jupiter's atmosphere and its relationship to moist convection remain unclear because it is difficult to observe the structure under the extensive surface cloud layer by remote sensing. We developed a two-dimensional dynamical model that incorporates condensation of H_2O and NH_3 and production reaction of NH_4SH (http://www.gfd-dennou.org/library/deepconv/). We ran the model for a long simulation time in order to examine a structure of moist convection in Jupiter's atmosphere that is established through a large number of life cycles of convective cloud elements. In this paper, we present the dependency of vertical cloud structure and convective motion on the abundances of condensible volatiles. Some suggestions for future probe missions will be given.

Model: The basic equation of the model is based on quasi-compressible system (Klemp and Wilhelmson, 1978). The cloud microphysics is implemented by using the warm rain bulk parameterization (Kessler, 1969) that is widely used in the modeling studies of the terrestrial clouds. The domain extends 300 km (30 bar -- 0.001 bar) in the vertical direction and 512 km in the horizontal direction. Radiative forcing is given as a function of height. To activate moist convection, strong radiation cooling is given. The atmosphere is cooled between 140 km (2 bar) and 200 km (0.1 bar) at a constant rate of 1 K/day. The abundances of condensible volatiles used in the each calculations are taken at 0.1, 1, 5, and 10 times solar.

Results: The results of the numerical simulations show that H_2O and NH_4SH cloud particles are advected above the NH_3 condensation level when the abundances of condensible volatiles are larger than 1 times solar. In the cases with larger abundances of condensible volatiles, moist convection develops with distinct temporal intermittency. In the period of active cloud development, the vertical cloud structure is similar to that obtained by 1 times solar. In the quiet period, however, two separate cloud layer forms: the lower one consists of H_2O and NH_4SH cloud particles, and the higher one is composed only of NH_3 cloud layer. These characteristics of the vertical cloud structures obtained by the numerical simulations are distinctly different from the classical three layer structure that has been expected by using equilibrium cloud condensation model. When the abundances of condensible volatiles are larger than 1 times solar, the H_2O condensation level acts as a dynamical boundary in the structure of convection. On the other hands, the NH_3 condensation level and the NH_4SH reaction level don't act as such dynamical boundary. When the abundances of condensible volatiles are taken at 0.1 times solar, however, even the H_2O condensation level doesn't act as a dynamical boundary: the downdraft can bring dry air from the tropopause to several ten bars level.

Suggestions for future Jupiter probe mission: We predict that, if probe descend the region of active moist convection, various mixtures of cloud particles of NH_3, NH_4SH, and H_2O will be encountered. Instrument that can determine not only the amount but also the composition of cloud and precipitation particles should be developed. The measurement of the composition, concentration, and size distribution of aerosol particles is also highly desirable. These informations are indispensable for improvement of cloud microphysical modeling. The structure and the dynamics of the atmosphere strongly depends on the deep tropospheric mixing ratios of condensable volatiles, especially water vapor. Because the Galileo probe failed to observe the representative composition of deep atmosphere, further direct measurements should be planned.

Poster session VI :
Entry, Descent, and Landing Technologies for Planetary Missions
(N. Cheatwood, D. Lebleu)

Poster 6.1	**J. Lachaud** *“A new experimental/theoretical approach to measure and extrapolate the ablative properties of TPS materials”*
Poster 6.2	**K. G. Medlock and J. M. Longuski** *“Aerocapture Ballutes for the Exploration of the Solar System”*
Poster 6.3	**I. Dupzyk** *“Current status of design, construction, and testing of supersonic parachutes for Mars EDL applications”*
Poster 6.4	**M. Capuano** *“ExoMars descent module EDL simulator”*
Poster 6.5	**B. Van Ootegem, D. Conte, J.M. Bouilly** *“Improvements on aerothermal flow characterization of industrial facilities”*
Poster 6.6	**J. McKinney** *“Mars Guided Parachute Guidance System for Wind Drift Compensation”*
Poster 6.7	**D. Pirotais, I. Montois, D. Conte, N. Sauvage, O. Chazot** *“Overview of dust effects during Mars atmospheric entries : models, facilities and design tools”*
Poster 6.8	**C. L. Tanner** *“Optimization of Earth Flight Test Trajectories to Qualify Parachutes for Use on Mars”*
Poster 6.9	**J. Benito and K. D. Mease** *“Mars Entry Guidance for High Elevation Landing”*
Poster 6.10	**A. Sanz-Andres** *“Pararrotors for planetary atmosphere exploration”*
Poster 6.11	**J. Benton, N. Bialke, S. Bradburn, L. Garbowski, R. Lane, G. Manfull, M. Murbach** *“A small, high velocity reentry probe capable of reconstructing atmospheric parameters”*
Poster 6.12	**P. Arfi** *“Safeto-Mars a multi-body simulation for planetary entry, descent and landing”*
Poster 6.13	**M. Murbach** *“The SOAREX-VI Re-entry Flight Test Experiment”*
Poster 6.14	**E. Millour** *“The New Mars Climate Database”*
Poster 6.15	**M. Paton** *“Atmospheric modeling for realistic EDL scenarios”*
Poster 6.16	**A. Verges, R. Braun** *“Analysis of the Mars Pathfinder Parachute Drag Coefficient”*

A NEW EXPERIMENTAL/THEORETICAL APPROACH TO MEASURE AND EXTRAPOLATE THE ABLATIVE PROPERTIES OF TPS MATERIALS

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In many applications, the design of thermal protection systems (TPS) relies on efficient thermostructural materials, such as Carbon/Carbon (C/C) composites. During re-entry, their walls undergo a surface recession, called ablation, mainly due to some gasification phenomena (namely oxidation and sublimation). This work is a contribution to the improvement of the understanding of material/environment interaction in order to :

- Enable a more reliable extrapolation of the models used for the design of TPS;
- Furnish guidelines for the fabrication of more efficient thermostructural materials.

The focus is set on the composite behavior which can be described through an average recession velocity and a surface roughness onset mainly caused by its heterogeneous anisotropic structure. In order to analyze this behavior, some original experiments have been carried out in an oxidation reactor in LCTS. Mass loss rate measurement and surface roughness analysis (SEM, microtomography) have shown that the composites featured a complex multi scale behavior. To explain these observations, a multiscale modeling strategy has been set up; it follows the characteristic scales of the composites: microscopic (fiber, inter-fiber matrix), mesoscopic (yarn, inter-yarn matrix), and macroscopic (homogenized composite) scales. The proposed models notably integrate the local recession of the wall (Hamilton-Jacobi equation), the heterogeneous gasification reactions, and mass transfer. A numerical simulation tool, based on Random Walks, has been implemented to solve these models. Using as a starting point the fibers properties (experimentally measured), the properties of the composite are inferred through two changing of scales. Numerical and analytical micro/macro models enable to predict and understand the macroscopic behavior of the materials in steady state and in transient regime. The theoretical results are in correct agreement with oxidation experiments [1].

The mathematical structure of the models and their morphological results are shown to be the same when the cause of ablation is either sublimation or oxidation [2]. This has been confirmed by comparison of surface roughness of samples ablated either by oxidation or sublimation.

Hence, the theoretical results validated in the case of oxidation can help understand ablation of C/C composites in plasma-jet tests or real flight experiments. It is namely possible to deduce by inverse analysis the intrinsic properties of the composite components in any condition from the analysis of the ablated surface morphology. Then, from the knowledge of the component individual properties and using the micro/macro models, the composite overall behavior can be extrapolated to different experimental conditions with an improved reliability. This method is here applied to C/C composites ablated using a plasma jet test (VKI) and a radiative device (CEA-CESTA) that leads to pure sublimation. As far as the material improvement is concerned, at low temperature, a weak interface between fibers and matrix is shown to initiate a weakest-link process and should be improved. At high temperature, a mass transfer limitation occurs so that the fibers tend to protect this weak phase, the fabrication effort should then be given to fibers improvement.

As a conclusion, there is obviously no ideal material, but the developed approach may help to choose or fabricate with an improved reliability the more appropriate candidate for a target application and to improve the near-wall phenomenological models for TPS design.

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AEROCAPTURE BALLUTES FOR THE EXPLORATION OF THE SOLAR SYSTEM

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Previous literature indicates that aerocapture with ballutes may provide significant mass advantages over heat-shielded probes. The reason for this advantage is that if the area-to-mass ratio of the ballute is large enough the heat flux can be made arbitrarily small. We consider the possibility of using ballutes in the exploration of the atmosphere-bearing bodies in the solar system, including Venus, Earth, Mars, Jupiter, Saturn, Uranus, Neptune, and Titan. We compare the mass cost of ballute aerocapture with propulsive capture and with previous work in which tethers are used for aerocapture in solar system exploration. Although both the ballute and tether technologies require further analysis in the areas of structures, materials, and guidance, both systems offer attractive advantages over aerocapture with aeroshells and over propulsive capture.

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CURRENT STATUS OF DESIGN, CONSTRUCTION, AND TESTING OF SUPERSONIC PARACHUTES FOR MARS EDL APPLICATIONS

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The delivery of payloads to high altitude challenge sites or the delivery of heavy payloads to the surface of Mars is a strong motivator for the development of parachute decelerator systems capable of deploying at Mach 3 or above. In this study the design space was explored for a decelerator system to be deployed between a flight Mach number of 2 to 3 using assumptions and simplified correlations for canopy material thickness, canopy mass, and steady-state descent velocity. The system coupling is preserved in the design of space exploration analysis performed in this study which is crucial to the understanding of highly complicated systems. The key design variables considered for the system analysis include canopy diameter, canopy drag coefficient, and free-stream Mach number at deployment. The objective of this work was to optimize the configuration of these variables based on the assumptions made. The crucial design variables were parametrically varied to examine main and interaction effects on the landing system design objectives and their coupling. A critical part of the ongoing research is to better understand the key physical phenomena that characterize the parachute system performance. Improved understanding of the parachute system will enable the effective optimization of the entry system as a whole. This will have a direct impact on the payload mass fraction delivered to the surface while also improving the performance and reliability of the Lander system.

An initial canopy configuration has been constructed and tested in drop tests to examine the low speed stability and performance of the system. The canopy configuration is a hybrid of disk-gap-band (DGB) and ring slot styles, each chosen for their favorable performance characteristics. At low speeds the canopy appears very stable, exhibiting negligible oscillation during its descent. This current design will be refined for improved performance at its designed flight conditions. Experimental testing will also be extended to include testing at higher dynamic pressures and at supersonic velocities.

A commercial Computational Fluid Dynamics code by ESI corp. was implemented to study fluid structure interaction effects and unsteady flow phenomena. Modeling was used throughout the design process to identify trends in the parachute system and performance. These solutions were benchmarked against drop tests performed with a constructed model and with data from past missions. Traceability studies were performed to validate the accuracy of the modeled systems. In addition, CFD solutions were used to derive scaling relations.

From this study it was determined that while some of the system coupling was initially maintained, other key design variables such as the entry probe's ballistic coefficient and its affect on the trajectory were also influential. The ballistic coefficient was shown to strongly affect the deployment Mach number. This is crucial for deployment in thin atmospheres, like that of Mars, where the entry system may not have sufficient time to reach a feasible deployment Mach number before impact.

EXOMARS DESCENT MODULE
ENTRY DESCENT LANDING SIMULATOR

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ABSTRACT

The Exomars mission is the first **ESA** led robotic mission of the Aurora Programme and combines technology development with investigations of major scientific interest. Italy is by far the major contributor to the mission through the strong support of the Italian Space Agency **ASI**.

ExoMars will search for traces of past and present life, characterize the Mars geochemistry and water distribution, improve the knowledge of the Mars environment and geophysics, and identify possible surface hazards to future human exploration missions.
ExoMars will also validate the technology for safe Entry, Descent and Landing (EDL) of a large size Descent Module (DM) carrying a Rover with medium range surface mobility and the access to subsurface.

The Exomars project is presently undergoing its Phase B1 with **Alcatel Alenia Space-Italia** as Industrial Prime Contractor. Additionally as Descent Module responsible, a dedicated simulation tool is under development in Alcatel Alenia Space-Italia, Turin site, for the end-to-end design and validation / verification of the DM Entry Descent and Landing.

Spacecraft Architecture
The Descent Module is a blunt-shape probe separated from either the Exomars Carrier Module (CM) or the Exomars Orbiter Module (OM) according to the launcher that would be finally selected for launch the Exomars Spacecraft Composite. So far launches with Soyuz 2b and Ariane 5 from Kuorou have been deeply investigated.

The DM is currently designed to land on Mars with two alternative airbags technologies:
- bouncing and rolling airbags (MER like)
- dead bed airbags (vented at touchdown to avoid overturning of the lander)

Entry Descent and Landing End-to End Analysis & Simulation

The execution of EDL End to End analysis is a driver for the consolidation of the Descent Module GNC control laws design, the definition of hardware performances and the triggering of the EDL events. Relying on one integrated simulation and analysis environment is needed to achieve robustness and reliability of a mission critical phase of the Exomars mission.
In order to support the EDL end-to-end mission definition and performances assessment, **the development of an EDL/GNC software simulator,** is started with the harmonization of the in-house available tools with the objective to build a simulator with the following main features:
Configurable: Vented /Non vented
Redefinition of mass properties following the main separation events
Modular: to support all EDL scenarios
Real time for HW in the Loop Test
Failure injection capability
Montecarlo, worst case and sensitivity analyses
Compatible with various environmental models
Includes models of sensors, actuators, CDMU and GNC algorithm, all can be replaced by HW

The architecture of the EDL/GNC simulator is presented in Figure 1.

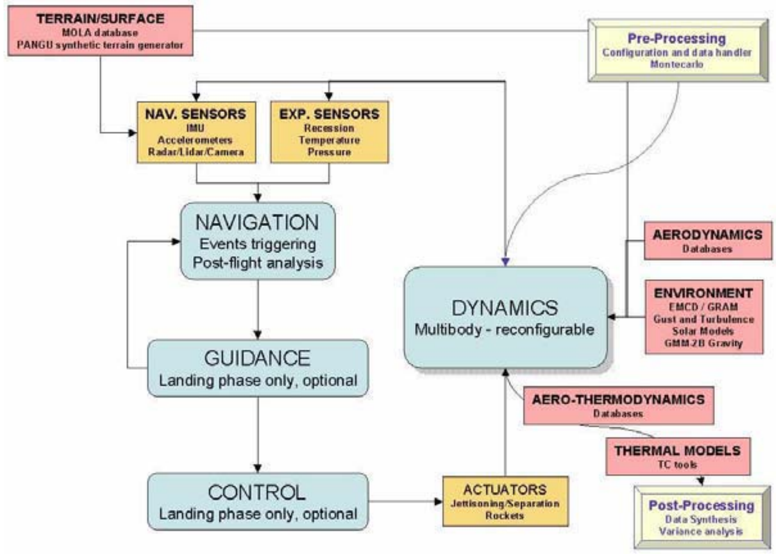

Figure 1: Architecture of the Exomars DM EDL/GNC Simulator

It is designed modular as much as possible to take care of the needed improvement / evolution in Phase B2/C/D and to allow efficient and smooth incorporation of specific models developed by Exomars lower tier subcontractors. Furthermore, the architecture will be consistently kept in line with the possible implementation of hardware in the loop tests, supporting the AIV phases.

The reference EDL reference scenario is designed to analyse the following phases:

1) DM Coasting
2) Hypersonic entry phase
3) Parachute Phase (Descent)
4) Active Thrusting Control Phase
5) Landing (airbags)

Each phase is separated by specific events that will be detected and triggered by the navigation and control functions like:

1) DM/CM Separation
2) Entry Interface (sensible atmosphere, conventionally 120÷125 km)
3) Parachute Triggering (altitude-dynamic pressure, reference Mach number)
4) Front Shield Separation
5) GNC Sensor package activation
6) Retrorockets Ignition and Control
7) Lander deployment/delivery.

IMPROVEMENTS ON AEROTHERMAL FLOW CHARACTERIZATION OF INDUSTRIAL FACILITIES

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ABSTRACT

The aim of the present communication is to introduce EADS ASTRIUM work in the field of aerothermal flow characterization.

ASTRIUM has a long experience of atmosphere re-entry simulation in order to investigate material behaviour of thermal protection system (TPS) of various space vehicles. Several plasma arc heaters test facilities of large power are commonly operated in the Aerothermal Laboratory located in the Aquitaine plant, near Bordeaux, France.

In order to meet new testing requirements, to extend its simulation capabilities for re-entry and to consider better quality test conditions, ASTRIUM has decided to adapt or to develop experimental measurements.

This article will focus on the main results of different flows concerning:

- *Heatflux sensors (mainly for high flux up to 60 MW/m^2)*
- *Molecules densities measurements (RAMAN and absorption) for N_2, O_2, NO, CO and CO_2.*
- *Characterization of a seeded plasma.*

Mars Guided Parachute Guidance System for Wind Drift Compensation
For IPPW-5 Session VI (Neil Cheatwood)

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Abstract

The Mars Guided Parachute (MGP) program is a portion of the current JPL Mars Technology Program and a portion of the Advanced Entry Descent and Landing Systems (EDLS) segment of that program. This paper provides an overview of the progress to date in the MGP program and a preliminary discussion of the final guided parachute flight test completed late in 2006.

As robotic and eventually human exploration of Mars continues to expand, the ability to place payloads on the surface of Mars, with increased precision is required. Science packages begin to investigate particular sites and features. Future robotic missions may combine the capabilities of multiple entry systems requiring close proximity landing. For instance, a sample return mission might separate entry vehicles for the sample gathering and sample return capabilities. Human Exploration will likely require crewed entry vehicles to land in close proximity to habitats and supplies. For these reasons the Mars Technology Program is working on greatly reducing the entry errors throughout the EDLS system. Other programs are investigating controlling errors during hypersonic entry, through lift vector rotation.

Similarly, other teams are investigating methods for enhanced precision in the final moments of landing, including additional features to provide a level of hazard avoidance.

Parachute flight, while brief remains a vital portion if the EDLS sequence at Mars, Earth and planets with a significant atmosphere, at least for most re-entry vehicles. Therefore, a level of control during this portion of the Descent phase can provide a significant improvement in landing precision. Guidance, Navigation and Control (GN&C) can compensate for significant errors in knowledge of atmospheric variables such as winds velocity and atmospheric density. To a lesser extent, GN&C can also compensate for

errors in the initial condition for the parachute descent phase that are due to similar atmospheric errors in the previous EDLS phase, such as the heat shield entry flight.

A number of Earth borne guided parachute systems are either available or being developed for a wide range of military cargo applications. These cover a range of weights from tens of pounds to tens of thousands of pounds. The current approaches include a variety of parachute systems from guidance of traditional cargo class parachutes, parafoils for very large cargo systems (30-40 Klb) and high speed flight of the cargo transitioning to a last minute, un-controlled parachute landing.

In the Mars environment, there have been only a handful of missions and perhaps 2-4 handfuls of development tests spanning the past 40 years. The reason for this is directly related to the cost of the testing. The best Earth bound test of a Mars relevant system requires delivery of the payload to an altitude of approximated 35 KM (115 Kft). This payload is typically balloon delivered. However, if the correct entry velocity is also required, depending on the parachute system, a rocket acceleration stage may also be required. For full system demonstration the correct shape of the entry vehicle may be required, further complicating (and raising the expense) of the testing.

The MGP program has much more modest goals. Our primary mission is to:

1) Demonstrate that a parachute can be actively guided in a Mars relevant environment
2) Demonstrate, through simulation, the value of such a control feature for future Mars missions.

To date, all major program goals and test objectives have been obtained, we have competed a single flight test of an unguided Mars Parachute in the relevant (earth high altitude) environment using a high altitude balloon drop. And in September 2006 another high altitude test of 2 Mars Beagle2 parachutes using closed loop guidance was accomplished.

This paper will present the results of the first flight test, a successful test with rather surprising results. We will also review the background of selecting the parachute and control system for the closed loop guidance test. And briefly discuss the successful results of the closed loop guidance tests. Also preliminary results of a Mars guided parachute landing simulation using parameters derived from the drop tests will be presented.

OVERVIEW OF DUST EFFECTS DURING MARS ATMOSPHERIC ENTRIES : MODELS, FACILITIES AND DESIGN TOOLS

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Introduction: . Mars atmosphere periodically witnesses dust storms that may leave solid particles in suspension for a long time. According to their high velocities, when reentry vehicles encounter such dust clouds even small particles can induce significant damage to the TPS, due to the high kinetic energies involved.
Charred layer erosion is the most evident effect of dust particles impacts, but other effects are present that need modelling for a comprehensive assessment of dusty flows

Dust effects review : in order to make the full assessment of dust effects on TPS, several issues have to be addressed (see figure at the end of abstract):

Step I : particles description : concentration (usually defined in g/m3), velocity (vehicle velocity), particles nature density and size (or size distribution)..

Step II : shock/particle interaction : this very brief event may be of importance in the next steps. If the shock is strong enough, and if the particles have low mechanical characteristics (water drops, brittle material), this shock crossing can induce particles breakup, which can influence subsequent slowing down

Step III : shock layer crossing : a good knowledge of particles deceleration is very important for the subsequent response of the TPS (which at first order depends on the particles kinetic energy). Other effects occur that need to be addressed : particles energy and momentum dissipation in the surrounding flow, particles heating and heat radiation to the heatshield, vaporization or sublimation,

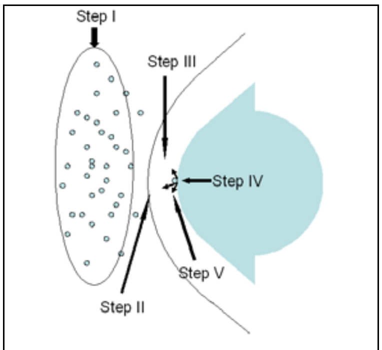
Step IV : particles/TPS interaction : This is the most visible consequence of dust presence on the vehicle entry. With the high velocities considered, the particles can erode ten times and much over their own mass which can give significant TPS recession.

More indirect effect is the roughening of the wall. If this roughening is sufficiently important, premature transition can occur in the boundary layer thus increasing the heat load to withstand.

Other effects than mere transition tripping are suspected to induce heat fluxes augmentation as high as 2 times the clear air heat fluxes (and supposedly over). These fluxes augmentation are the second known effect in importance (after mechanical erosion) that dusty flows induce over reentry vehicles.

Step V : debris : after impact, many debris are ejected back in the flow. The subsequent history of these debris is interesting to assess, as they can seed the boundary layer and modify its behavior. Eventually, incident particles and debris can have an influence on payload bio-contamination in rear flow.

Paper outline : the paper will propose an overview of these effects within the frame of Mars entries, with a focus on step III and IV which are the most important issues as far as heat fluxes and material response are concerned.
Existing models, design tools and facilities will be reviewed, highlighting limitations and expected improvements. These involve material characterization, dust/flow interactions assessment and numerical codes.



Specific issues due to dust presence in hypersonic flows around probes

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OPTIMIZATION OF EARTH FLIGHT TEST TRAJECTORIES TO QUALIFY PARACHUTES FOR USE ON MARS

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Introduction: It is challenging to replicate critical flight conditions experienced on Mars during entry, descent, and landing (EDL) with flight tests on Earth. Differences between Earth and Mars in their atmospheric density and composition, speed of sound, and the acceleration due to gravity make it difficult to simultaneously create the proper combination of trajectory parameters. Thus, Earth-based flight tests must be carefully designed to best simulate the anticipated flight conditions at Mars in order to properly qualify an EDL system.

Abstract: This paper presents the development and implementation of a simulation designed to optimize flight test trajectories through the Earth's atmosphere. These flight test trajectories attempt to replicate the conditions experienced by a parachute during supersonic descent through the Martian atmosphere. Trajectory simulation is accomplished by integrating the planar equations of motion over a flat planet. Over the pertinent portion of the EDL trajectory, the most critical parameters that need to be matched in a supersonic flight test are the peak opening load and the parachute load time history. The parachute's peak opening loads are calculated using two different models: an empirical inflation curve model and an apparent mass model. To demonstrate the simulation, an example mission of a large payload descent at Mars is considered which has a mass of 4,000 kg at parachute deployment and uses a 30 m nominal diameter disk-gap-band parachute. This payload-parachute system is used to generate a set of reference descent trajectories through the Martian atmosphere. Optimization of the Earth flight test trajectories is accomplished by minimizing the difference in the parachute load time-history between the reference Mars trajectory and the Earth flight test trajectory. The optimizer is constrained to match the peak opening load and is allowed to vary the Earth flight test payload mass, initial altitude, and initial flight path angle. The results of the flight test trajectory optimization and the practical considerations of implementing such a test are discussed. Aerodynamic heating of the parachute, suspension line elasticity effects, and how they impact the Earth flight tests are also briefly considered.

MARS ENTRY GUIDANCE FOR HIGH ELEVATION LANDING

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Introduction:

A desired capability driving Mars entry, descent and landing technology development is landing at higher elevation sites without compromising a horizontal accuracy of 10 km. Mars landings to date have been at sites with elevations below -1.3 km MOLA (Mars Orbiter Laser Altimeter). To reach much of the ancient highlands, a majority of the southern hemisphere, requires landing at elevations as high as +3 km MOLA. The entry guidance task is to deliver the lander accurately to the supersonic parachute deployment point within the parachute altitude and Mach number constraints. The higher the required landing site elevation is, the higher the minimum acceptable supersonic parachute deployment altitude is and the lower the atmospheric density is that the lander must maneuver in during the final stage of entry. In previous work, an entry guidance algorithm called EAGLE, for Evolved Acceleration Guidance Logic for Entry, was developed. It consists of a trajectory planner and a trajectory tracker.

Trajectory planning:

In this work, modifications to the planner are developed to allow guidance to higher final altitudes. The new trajectory planning strategy is based on insight from examining drag profiles for trajectories that maximize deployment altitude. An optimization program was used to generate entry trajectories for different required downranges and maximum parachute deployment altitudes. The planner determines a reference drag acceleration profile to match the required downrange and plans one or two bank reversals to match the required crossrange, while delivering the vehicle at high altitude.

Trajectory tracking:

The tracking law tracks the reference drag profile by commanding bank angle, hence tracking a downrange vs. energy profile. In the presence of atmospheric and aerodynamic modeling errors large parachute deployment crossrange and altitude errors can arise. To correct for this, a final phase is introduced from Mach 11 until the parachute deployment that drives the heading error to zero while shaping the transient to control the parachute deployment altitude.

Simulations and Results:

Simulations show that the cobined planning and tracking laws can deliver a lander to a higher parachute deployment site without comproimising the horizontal accuracy in the presence of large atmospheric and aerodynamic uncertainties.

References:

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2. Leavitt, J. A. and Mease, K. D., Feasible Trajectory Generation for Atmospheric Entry Guidance, J. Guidance, Control, and Dynamics, Vol. 30, No. 2, pp. 473-481, 2007.
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PARARROTORS FOR PLANETARY ATMOSPHERE EXPLORATION.

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Introduction: The aim of this paper is to present some results of the study performed concerning the dynamics of pararotors. This type of deceleration systems is a good candidate as a probe to get atmospheric data of a planet with enough dense atmospheres.

The results summarized here are those obtained in some recent research projects concerning the aerodynamics modelization, the stability analysis and vertical wind tunnel tests. The aerodynamic model allows to perform a preliminary sizing of the probe similar to the ones considered in the model development. The aerodynamics model is based on wind tunnel experimental tests performed with models supported by means of anchored rotation axis. The stability study gives information relevant to the mass distribution allocation inside the probe, in order to obtain a stable flight. It is shown that the stability map can be defined in terms of the ratios of moments of inertia, but some variations can be found if the pararotor blades are unsymmetrically tilted. Some experimental results obtained with almost free flying models in a vertical wind tunnel are reported, aiming to validate the theoretical models.

Although the pararotor type considered here consist of a cylinder with two opposite blades of small aspect ratio, the results involving aerodynamic tests can be extended straightforwardly to other similar shapes. For dissimilar shapes, new tests would be needed to determine the aerodynamic coefficients which appear in the theoretical model. However, the stability model can be applied to a general shape to determine their stability just from the values of the principal moments of inertia.

POSTER: A SMALL, HIGH VELOCITY REENTRY PROBE CAPABLE OF RECONSTRUCTING ATMOSPHERIC PARAMETERS

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Abstract: In order to provide NASA Sub-Orbital Aerodynamic Re-Entry Experiments (SOAREX) with atmospheric conditions for trajectory analysis, we have designed and built a small atmospheric entry probe as an undergraduate senior design project. Previous planetary entry missions such as Mars Pathfinder and Huygens have used accelerometry to derive atmospheric density, from which pressure and temperature can be derived (Withers 2003, Kazeiminejad 2006). By using an entry vehicle with a well-characterized drag coefficient, atmospheric parameters derived from accelerometry can also provide an accurate reconstruction of the upper atmosphere. An off-the-shelf aerospace radar transponder was used to provide position (and hence, velocity and acceleration) accurately while occupying a small space and drawing low power. A full radar transponder package with redundant support systems to be used for atmospheric reconstruction has been designed to fit within a 20 cm diameter sphere. The probe is able to survive Earth atmospheric entry at 4 km/sec from an altitude of 500 km with a polytetrafluoroethylene (Teflon®) ablative thermal protection system. We will discuss the construction of the entry probe, the atmospheric reconstruction process, and the applicability of small entry vehicles to atmospheric experiments.

Bibliography: P. Withers, et al, "Analysis of entry accelerometer data: A case study of Mars Pathfinder," *Planet. Sp. Sci.*, vol 51, p. 541-561, 2003.

Kazeiminejad, B., D.H. Atkinson, et al. "Huygens' Entry and Descent through Titan's Atmosphere - Methodology and Results of the Trajectory Reconstruction", accepted for publication in *Planet. Sp. Sci,* 2006.

SAFETO-MARS A MULTI-BODY SIMULATION FOR PLANETARY ENTRY, DESCENT AND LANDING

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In the frame of the Aurora Exploration Program, ESA initiated industrial studies for the ExoMars mission. Alcatel Alenia Space-France led consortium was awarded the Phase B1 Entry Descent and Landing Systems studies under Alcatel Alenia Space Italy mission prime. The primary objective of Aurora is to create, and then implement, a European long-term plan for the robotic and human exploration of the solar system, with Mars, the Moon and the asteroids as the most likely targets. First mission of the Aurora Exploration Program, ExoMars is scheduled for a 2013 launch. It has the technical objective to demonstrate critical technologies linked to Mars mission, including the critical Entry Descent and Landing phases and the scientific objective at establishing whether life ever existed or is still present on Mars. The EDLS functions are to ensure the Entry, Descent and Landing phases of the Descent Module. This module is housing a Rover carrying an exobiology payload and ensures its safe delivery on the Mars surface.

Based on AAS-F reliance on simulation for the Huygens mission, the AAS-F Entry Descent and Landing System team has merged and significantly upgraded the various Huygens, Mars Sample Return (MSR) and NetLander tools initially developed for subsystems designed independently. The adaptation and merging of these tools led to an accurate and detailed end-to-end EDL trajectory modelling and simulation tool : Simulation Adapted For End to end Trajectory Optimisation- MARS (*SAFETO-MARS*). The multi-body capability of this tool has been extensively used as the major simulation tool during the ExoMars Phase A for EDLS sizing and sequence optimisation. The EDLS team was then able to build more robust sub-systems assembled in an optimised EDLS.

The entry, descent and landing phases of the planetary mission consists of a precise sequence of events : atmospheric entry, parachutes deployment, frontshield separation and a terminal descent phase based on a proulsion reaction control system. Using SAFETO-MARS this sequence is accurately modeled. In addition, SAFETO-MARS enables the user to obtain a detailed physical insight and sensitivity analyses of the phenomena involved for each subsystem.

The entry phase simulation is based on a 6-DOF flight simulator developed for MSR and validated in the frame of the ESA robust control study. The descent phase simulation is built upon a multi-body parachute trajectory simulation. The parachute and the suspended bodies are treated as 6 Degree-of-Freedom (6 DOF) bodies for improved accuracy. Different modeling of the Reaction Control System (fixed and modulated thrust) are implemented. The RCS module is an enhanced version of the MSR simulator. The tool, developed on Matlab, is coupled to the EMCD data base, complies with the atmosphere modelling requirements for MARS missions including winds or gust effects, and provides a wide range of output data and graphics.A Monte Carlo dispersion analysis is available to statistically assess the robustness of the EDL sequence to off-nominal operational conditions and environments to assure that all EDLS requirements and engineering constraints are satisfied.

THE SOAREX-VI RE-ENTRY FLIGHT TEST EXPERIMENT

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ABSTRACT

The experiment definition and development of the SOAREX VI (Sub-Orbital Aerodynamic Re-entry EXperiment) flight test will be discussed. Scheduled for launch on the ATK X1 sub-orbital flight during 4th quarter of 2007, this flight in the SOAREX series is a mission of opportunity- class payload that will permit one or more re-entry bodies to enter the atmosphere at approximately 4 km/s. The trajectory that is provided presents certain challenges, in that the launch site is the Wallops Flight Facility with a down-range splash point several hundred km north of Antigua (total downrange distance of 2037km/1100NM). These included unique telemetry system choices and means of collecting accurate entry trajectory data due to the distance from any available land range. The principal entry experiment is the SCRAMP (Slotted Compression RAMP) probe, which is uniquely self-stabilizing and permits a simplification of the interface to the upper-stage/payload ejector system. The sensor system will permit measurement of the key aerodynamic parameters, as well as of the shock-shock interaction which occurs on the outer radius of the SCRAMP aft flare (the principal drag and stabilization element). The data from this experiment will be used to improve the understanding of this shock-shock behavior, provide valuable design data for this class of highly stable probe being used for planetary missions, and finally, streamline a capability of conducting comparatively inexpensive flight experiments at long down-range distances.

THE NEW MARS CLIMATE DATABASE (VERSION 4.2)

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Introduction: The Mars Climate Database (MCD) is a database of meteorological statistics compiled from General Circulation Model (GCM) numerical simulations of the Martian atmosphere. The GCM has been developed over the years at Laboratoire de Météorologie Dynamique (Paris, France) in collaboration with The Open University (Milton Keynes, UK), the Oxford University (Oxford, UK) and the Instituto de Astrofísica de Andalucía (Granada, Spain) with support from the Centre National d'Etudes Spaciales (CNES, France) and the European Space Agency (ESA). The MCD has been validated using many sets of available observational data and is intended to be useful in the framework of engineering applications such as atmospheric trajectory computations as well as in the context of scientific studies. The new version of the MCD includes all the features of its predecessors (with some improvements), and some additions such as a new "high resolution mode" and extra statistics on the altitude-wise standard deviations of main atmospheric variables.

Overview of data provided in MCD v4.2: The database extends up to ~350km (i.e., up to and including the thermosphere) and provides statistics on atmospheric temperature, density, pressure and winds along with radiative fluxes as well as atmospheric composition (including dust and water vapor and ice contents) since our GCM includes both full water and chemistry models. In order to represent the full range of atmospheric dust distribution and solar Extreme Ultra Violet (EUV) inputs on Mars, 8 different "dust and EUV" scenarios are considered.

In addition to mean values of meteorological variables, the MCD includes the variability thereof in various ways: Firstly, standard deviations of main variables are supplied (in two different contexts, either pressure-wise, as in previous versions of the MCD, or altitude-wise). Secondly, users may reconstruct the variability of the atmosphere by adding perturbations to mean values, either in the form of large scale perturbations, using Empirical Orthogonal Functions (EOF) derived from GCM runs, or small scale perturbations, by adding gravity waves of user-defined wavelength.

As of this new version, MCD access software now includes the implementation of a "high resolution mode" which combines MCD data with high resolution (32 pixels/degree) MOLA topography and atmospheric mass correction from Viking Lander 1 pressure records to yield, within the restriction of the procedure, high resolution values of atmospheric variables.

Obtaining and using the MCD: The Mars Climate Database is freely available and distributed in two different forms: in a "reduced" version on the web, useful for moderate needs, at http://www-mars.lmd.jussieu.fr, and as a DVD-ROM (which contains the data files but also the access and post-processing software, in the form of a Fortran 77 subroutine; IDL, Matlab, Scilab, C and C++ interface examples are also included) for intensive and precise work.

ATMOSPHERIC MODELING FOR REALISTIC EDL SCENARIOS

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Introduction: With manned flights planned to Mars during the next decades it becomes crucial to control the landing vehicles' entry, descent and landing as precisely as possible. All of the current scenarios foresee the deployment of an unmanned insfrastructure before the manned landing takes place. A prerequisite for this approach is the possibility to constrain the area of touch-down of all mission vehicles to a manageable small area of only hundreds of meters in diameter. If large masses of fuel for navigating should be avoided, the atmospheric influences on the descent vehicles along the descent path have to be estimated as correctly as possible.

Modeling of Entry through the Martian atmosphere: Global models of the Martian atmosphere have been developed over the last decades and been compared with in-situ measurements from earlier and current Mars landing missions and with observations from orbiting remote sensing instruments. While they help to understand the development of large-scale phenomena they are not detailed enough to provide the information about possible atmospheric influences on the descent trajectory of a landing vehicle. Based on meteorological high-resoloution weather forcast models used by the Finnish and other European meteorological institutes, we developed a 3-D Mars Local Area Model (MLAM) to describe mesoscale developments of the Martian atmosphere.

Combining these results with boundary conditions used for the Aerobrake 2D program, high fidelity simulations for fine tuning EDLs are possible. Variations of temperature, atmospheric pressure and wind speed and direction as a function of altitude and ground topography can be used to optimize the entry scenario, shape of the entry vehicles and effective use of active guidance systems. A similar approach can be used for developing atmospheric re-entry scenarios into the Earth atmosphere for sample-return and manned missions.

Different examples of such iterative optimization steps will be shown, based on vehicles described in the Mars for Less mission (Bonin, 2006) and Apollo-6 re-entry analysis.

ANALYSIS OF THE MARS PATHFINDER PARACHUTE DRAG COEFFICIENT

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Introduction: The successful landing of the Mars Pathfinder (MPF) mission on July 4, 1997 included an entry, descent, and landing (EDL) scenario that used a disk-gap-band parachute for Lander deceleration. Flight reconstruction at NASA Langley Research Center and Jet Propulsion Laboratory of the entry using MPF flight accelerometer data concluded that the parachute decelerated faster than predicted. In the included investigation, the methodology used to estimate the MPF parachute drag coefficient is recreated and evaluated to validate the method used and to estimate the effect of additional terms in the equations of motion. Analysis of these effects characterizes the degree of underperformance of the MPF parachute leading to increased knowledge of parachute design for futue Mars landing missions, such as the Mars Phoenix Lander. This paper provides an explanation of the methodology used to estimate the MPF parachute drag coefficient and the effect of the acceleration term on the parachute drag coefficient.

References: Desai, P. N., Schofield, J. T., and Lisano, M. E., “Flight Reconstruction of the Mars Pathfinder Disk-Gap-Band Parachute Drag Coefficient” 17th Annual AIAA Aerodynamic Decelerator Systems Technology Conference and Seminar; Monterey, CA, 19-22 May 2003. AIAA-2003-2126.

Poster session VII : Emerging, Enabling, and Extreme Environment Technologies; Cross-Cutting Technologies
(L. Peltz)

Poster 7.1	**J. Founds** *“High-voltage series mosfet output driver in CMOS technologies for extreme environments”*
Poster 7.2	**V. Abidzina & al.** *“Effects of Glow-Discharge Plasma on Metals, Polymers and Glasses”*

Poster HIGH-VOLTAGE SERIES MOSFET OUTPUT DRIVER IN CMOS TECHNOLOGIES FOR EXTREME ENVIRONMENTS

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Abstract- This poster presents the stacked MOSFET as an integrated solution for digitally controlled drive actuators. The stacked MOSFET is a scalable implementation of series connected MOSFETs for high-voltage monolithic switching applications. A single low-voltage input signal activates the first MOSFET, which in turn causes the entire stack of devices to turn on by charge injection through parasitic and inserted capacitances. Voltage division provides static and dynamic protection by balancing the output voltage across the stack so that the voltage seen by each device never exceeds its break down voltage.

The stacked MOSFET's basic unit switches its nominal voltage rail to rail. As more of these basic units are added to the stack, the entire stack can switch a higher voltage rail to rail. For example, the nominal voltage for the 0.18μm process presented is 2.5 volts. A two device stack switches 5 volts rail-to-rail and a three device stack switches 7.5 volts rail-to-rail and so on. The stack is scaled up until it can switch the desired high voltage rail to rail. In theory the stacked MOSFET is *n* times scalable, but in practice the size of the stack is limited by the substrate break down voltage and the design accuracy of the parasitic and inserted capacitance.

The stacked MOSFET can be implemented in integrated CMOS topologies, which have clear advantages for actuator drive chips. The stacked MOSFET converts a logic level input signal into an output signal high enough for actuator drive requirements on a single chip. This minimizes the power consumption and mass of the actuator drive chip, which is important for atmospheric entry and descent probes where power and space are critical commodities. The stacked MOSFET has been successfully fabricated in radiation hard processes. For long duration missions such as interplanetary exploration, radiation hard components are an essential consideration.

Past work by Hess, Li, and Nelson [1] proved that a 0.8μm process (nominal 5V devices) stacked MOSFET could drive 80 volts, and when paralleled could output 1.2 amps continuously for a period of at least five months. A similar configuration was fabricated in a 0.18 μm process. Preliminary experimental tests match simulations for the 0.18 μm process, which shows the stack can switch 10 volts reliably. These results and the test performance under extreme temperatures will be presented at the conference.

[1] Hess, H.L., H. Li, and R. Nelson, "High Voltage Series MOSFET Configuration for Micropropulsion Application," NASA JPL System on a Chip Seminar, Pasadena, California, 11 February 2002.

Effects of Glow-Discharge Plasma on Metals, Polymers and Glasses.

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This work is focused on the investigation of glow-discharge plasma influence on several types of materials such as metals, polymers and glasses.

Instrumental steel, titanium alloys and hard alloys were chosen as metallic samples of the investigation. Glassy Polymeric Carbon (GPC) was chosen as polymer samples. GPC is used for the harsh environment of space as well as for protective coating against extreme environments such as high temperature, highly ionizing radiation, as well as corrosive environment. Silica glass with various coating was also used in this investigation.

Glow-discharge plasma treatment was carried out in a specially constructed plasma generator, where materials were irradiated by ions of residual gases in vacuum. The ion energy was 1-3 keV, while the current in the plasma generator was maintained at 30-70 mA. Barometric pressure of residual gases in the plasma generator chamber was 5.3 Pa. The irradiation fluence was maintained at $2 \cdot 10^{17}$ per cm^2. The temperature in the chamber was controlled during the irradiation process and did not exceed 323 K. The irradiation time varied from 10 to 180 minutes.

After plasma treatment, microhardness and hardness of metallic samples, their fine dislocation structures and electrical resistivity were investigated.

Chemical changes in GPC were studied using FTIR, micro-Raman spectroscopy and Rutherford Backscattering Spectrometry (RBS). Porosity changes were monitored through introducing lithium from a molten LiCl salt into GPC and using the (p,α) nuclear reaction analysis (NRA) to measure Li concentration in treated GPC.

Changes in glass properties were monitored by optical absorption photospectrometry and RBS.

Poster session VIII : Earth Entry, Descent and Landing (EDL) for Sample Return and Crewed Missions
(J. Arnold, B. Foing)

Poster 8.1	**G. Ferraro** *“Trajectory Optmization for an Atmospheric Re-Entry Test Bed”*
Poster 8.2	**R. Prakash and C. Dunn** *“An Altimeter and Velocimeter Trade Study for the Orion Terminal Descent System”*
Poster 8.3	**M. Ivanov, W. Strauss and R. Maddock** *“Entry, Descent and Landing Mission Design for the Crew Exploration Vehicle Thermal Protection System Flight Test”*
Poster 8.4	**S. Surzhikov and D. Kotov** *“Current Problems of Physical-Chemical Computational Fluid Dynamics Applied to Earth Re-Entry Space Vehicles”*
Poster 8.5	**C. Seybold, Ch. de Jong, C. Ong** *“Flight System Design for the Crew Exploration Vehicle Thermal Protection System Material -Qualification Flight Tests”*
Poster 8.6	**G. Vekinis and G. Xanthopoulou** *“Hybrid-TPS: Current Developments and Prospects”*

TRAJECTORY OPTIMIZATION FOR AN ATMOSPHERIC RE-ENTRY TEST BED

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The objective of this work is to provide the best configuration and performance of a National Spaceplane Demonstrator in front of a strategy of long re-entry from Sub-Orbital Mission: in particular the study is focused on the control problem of the Flight Test Bed (FTB-X) trajectory optimization. FTB-X is part of the Unmanned Space Vehicle (USV) program, carried out by the Italian Aerospace Research Center (CIRA). Its main goals are the development and the in-flight tests of the critical technologies for autonomous, fully reusable, winged body launch vehicles. In this framework the main goal of the demonstrator is the validation of the technological and operational aspects of the sub-orbital re-entry test mission.

The re-entry trajectory is established by using appropriate environmental models and a point-like spacecraft (pure center of mass translation). The guidance law has been defined in terms of angle of attack and no bank modulation has been taken into account. Mass, density and re-entry angle are also considered.

The cost function consists of maximizing the aerodynamic efficiency of the vehicle, which is expressed by the Lift-to-Drag ratio. The optimal control variable is constituted by the angle of attack. During the flight, the re-entry vehicle is subject to heating rate, overload and dynamic pressure constraints, which identify the flight entry corridor. An additional condition to be satisfied during the re-entry also applies: the flight path gradient must be always lower than a maximum admissible value.

The optimization is carried out using a MATLAB program, which relies on the fmincon function (MATLAB Optimization Toolbox) and on a Simulink model implementing the FTB-X dynamics. The optimal control problem is discretized into a nonlinear programming problem using the direct method. The angle of attack and the state variables are selected as optimal parameters in correspondence to fixed nodes, which are equally spaced in time (along the trajectory). The MATLAB fmincon function applies an iterative method that stops with the occurrence of the convergence of the optimization process.

It has been demonstrated that the algorithm is able to generate a feasible re-entry trajectory satisfying all the constraints with a reasonable CPU time. Additional tests have been defined and implemented in order to validate the effectiveness and robustness of the optimal solution.

An Altimeter and Velocimeter Trade Study for the Orion Terminal Descent System

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Orion is the next generation spacecraft that will reenter the Earth's atmosphere and safely land astronauts on the surface of the Earth. The terminal descent sequence of the Orion capsule is an essential series of events in the Orion mission timeline. Currently, three landing system architectures are being considered to attenuate the landing of the capsule: an airbag system, a retro-rocket system, and a hybrid airbag-rocket system. The airbag only and hybrid systems require that the heatshield be separated to expose the landing system, while the rocket only landing system uses blowout ports in the heatshield and retains the majority of the heatshield to act as the secondary attenuation device. For the two architectures that utilize retro-rockets, a method of sensing the horizontal and vertical velocities (velocimetry) as well as the altitude of the capsule above the surface of the Earth (altimetry) is imperative in order to trigger the retro-rockets to fire at a specific altitude. Neither velocimetry nor altimetry is needed for the airbag only architecture because the airbags do not require inflation at a specific altitude. From dynamics analysis, it was quickly determined that the use of retro-rockets on the landing system would require a very high performing altimeter and velocimeter. A trade study was conducted to determine if an altimeter and velocimeter currently exists or could be developed to meet the requirements.

Two sets of requirements were developed for the sensor architectures. One set of requirements was based on the Orion heatshield being separated prior to landing (the hybrid landing system), while the other set of requirements looked at the Orion heatshield being retained (the rocket only landing system). The retained heatshield architecture was more demanding from a sensor point of view because the rockets needed to slow the capsule down to 8 ft/s prior to landing, whereas in the separated heatshield architecture, the rockets only needed to slow the capsule down to 15 ft/s. The higher allowable landing velocity for the separated heatshield architecture is because the airbags provide more attenuation on landing than the heatshield. Lower landing velocities imply a tighter landing dispersion and more stringent sensor requirements. Both sets of requirements were derived from the currently defined Orion architecture and the flow down of the landing system requirements. These requirements include performance requirements such as altitude and velocity accuracy, environmental requirements such as the sensor operating temperature, and interface requirements such as specifying the interfaces between the sensor and onboard avionics components already existing on Orion. In addition, all architectures had to be dual fault tolerant to comply with higher-level Orion requirements.

In order to find an altimetry and velocimetry architecture that could best meet the requirements, a broad search of various altimetry technologies was conducted. The following technologies were researched: radar, laser, GPS/pseudolite, structured lighting, passive imaging, gamma ray, sonar, and mechanical sensors. The technologies were evaluated based on a set of discriminators that included each sensor's ability to meet the requirements. Dual fault tolerant architectures were then composed from one or more of the researched technologies and evaluated against each other based on a similar set of discriminators. This paper provides an in depth discussion of the steps taken in this trade study including an overview of the various technologies investigated, high-level design information for each architecture, the rationale supporting each architecture choice, and a comparison of the altimeter architectures to each other. Finally, the suggested architecture choice for the Orion altimetry and velocimetry system will be discussed.

POSTER - ENTRY, DESCENT, AND LANDING MISSION DESIGN FOR THE CREW EXPLORATION VEHICLE THERMAL PROECTION SYSTEM QUALIFICATION FLIGHT TEST

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Introduction: In support of NASA's Vision For Space Exploration, a new manned space vehicle (the Crew Exploration Vehicle or CEV) is being designed to replace the aging Space Shuttle fleet. Integral to the CEV development is the design of the systems necessary for the safe return of the crew back to Earth, especially from high speed lunar return missions. One of the many system challenges for a safe return is the design of the Thermal Protection System (TPS) necessary to protect the astronauts from the extreme thermal environments encountered during the Earth Entry, Descent, and Landing (EDL) operations. The TPS development activity along with other system development activities have spawned multiple initiatives internal to NASA to define the requirements for full scale and sub-scale flight tests required to validate and/or qualify these key systems for flight ready status. The Jet Propulsion Laboratory (JPL) was tasked to manage an initial study exploring the flight system development of a sub-scale CEV capsule for the specific purpose of supporting the qualification of the TPS material (PICA) for both low Earth orbit and lunar return missions. Integral to this flight system study was the EDL mission design necessary to achieve the aerothermal test requirements while minimizing the overall cost of the flight test.

EDL Mission Design: The EDL mission design process was challenged with the task of assuring that the flight environments experienced by the sub-scale CEV would encounter specific full-scale thermal conditions with very high accuracy. These thermal test conditions constituted a test "box" defined by ranges in total heat flux, radiative heat flux, surface pressure, and shear stress that the TPS material had to "fly" through at some point during the EDL event. In addition, environment limits on these parameters had to be accounted for as well as g-load for structural considerations. The official CEV aerothermal database was mined to discover vehicle angles of attack, mach numbers, and dynamic pressures that provided favorable thermal conditions from which to begin the detailed trajectory design process. A trajectory design optimization problem was formulated to achieve the thermal objectives while attempting to reduce launch vehicle costs via minimizing entry mass and velocity. In addition, Monte Carlo uncertainty analyses were performed to characterize the landing footprint as well as to assure that adequate margin was included in the nominal mission design resulting in a high probability of achieving the test conditions. Ultimately, two ballistic flight tests satisying all thermal objectives for a minimum achievable launch cost were identified in this study. These ballistic flight tests demonstrated a very large landing uncertainty ellipse necessitating the need for further study in applying on-board guidance in an attempt to reduce the landing ellipse while not compromising the thermal test objectives; therefore, synergy with other GNC flight tests seemed apparent and necessary. As a result, current activity is focused on the feasibility of combining these TPS test flights with other CEV test flights to 1) leverage the synergy of seemingly disparate flight objectives and 2) further reduce the overall cost of the entire CEV flight test program. Toward this end, mission design studies involving guided skip and non-skip entry trajectories that combine the testing of both the GNC systems and the TPS material are being explored.

Conference Proposal: This abstract proposes that a POSTER be presented at the conference outlining the entire EDL mission design process for the CEV TPS flight test study. This poster(s) would cover an overview of the 1) EDL mission design problem formulation utilizing the aerothermal database, 2) nominal and dispersed predicted flight test environments and landing footprints, and 3) alternative EDL mission design options currently under consideration.

CURRENT PROBLEMS OF PHYSICAL-CHEMICAL COMPUTATIONAL FLUID DYNAMICS APPLIED TO EARTH RE-ENTRY SPACE VEHICLES

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Several kinetic models of gas phase reactions have been studied with CFD code NERAT for interpretation of aerothermodynamics of space vehicles entering into the Earth atmosphere at hyperbolic velocities. Conditions of the FIRE-II flight experiment were considered. It is shown that studied kinetic models (the Park model, the Dunn and Kang model, and the Martin model) provide acceptable agreement between convective heat fluxes on space vehicle surface, but the models give different temperature fields that can be significant for prediction of radiative heating.

A number of future space missions include scenarios where radiative heating becomes important. To develop a prediction computational fluid dynamics (CFD) tool for re-entry flows where radiation is important, some major areas are addressed. Most significant of them are the following: (1) physical-chemical kinetics of high temperature dissociated and ionized gases, (2) transport properties of the gases mixtures, (3) spectral radiative properties of high temperature gases and low-temperature plasmas, (4) numerical simulation algorithms for prediction of non-equilibrium gas mixtures dynamics and radiation heat transfer in volumes of various geometry, (5) models of physical and chemical processes attendant interaction of gas flows and radiation with thermo-protection systems (TPS) of space vehicles (including their thermo-chemical destruction, ablation, sublimation, etc.).

Flight data of the Project Fire-II vehicle [1] is the one of the most documented experimental data, which can be used for verification of the CFD codes. The Project Fire-II experimental data has also two significant peculiarities. Namely, part of the measured data corresponds to conditions of local thermodynamic equilibrium (LTE) behind the front of shock wave, while some measured data corresponds to strongly non-equilibrium conditions. Just these experimental data are of great practical interest for development of modern directions of physical gas dynamics. Unfortunately, many attempts of CFD interpretation of the flight data showed significant discrepancy of the predicted and experimental data [2].

The general goal of the study is to analyse numerical simulation results on aerothermodynamics of super-orbital space vehicles, obtained by CFD code NERAT (**N**on-**E**quilibrium **R**adiation **A**ero-**T**hermodynamics) with different models of chemical kinetics of ionised air. This CFD model is based on equations of viscous, heat conducting, chemically non-equilibrium, emitting and absorbing gas, and includes databases of thermodynamic, kinetic and radiation properties. The CFD model uses its own model of physical-chemical kinetics, which is oriented to specific flight conditions of Earth entry. A special attention is attended to interpretation of Fire-II experimental data (velocities of 11.36-6.19 km/s at altitudes 76.42-37.19 km in the Earth atmosphere).

Code NERAT was used for different trajectory points counting for conditions of Fire-II experiment. Three kinetic models of high-temperature air [2-4] were used for theses calculations. It is shown that these kinetic models provide satisfactory agreement on convective heating only for equilibrium conditions. Using of different models of non-equilibrium dissociation and vibrational relaxation results in increasing of the differences between numerical predictions. So it seems that more precise, ab-initio kinetic models need to be developed.

Radiative heating of space-vehicle surface was predicted by code ASTEROID using multi-group spectral models of high-temperature air. Numerical studies of influence of the chosen number of spectral groups on total radiation heat flux is presented.

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POSTER: FLIGHT SYSTEM DESIGN FOR THE CREW EXPLORATION VEHICLE THERMAL PROTECTION SYSTEM MATERIAL QUALIFICATION FLIGHT TESTS

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To support qualification of Thermal Protection System (TPS) heatshield materials planned for NASA's *Orion* Crew Exploration Vehicle (CEV), a potential flight test program called TORCH - Testing of Re-entry Capsule Heatshield - was developed. For a test platform, the TORCH program designed a two-meter diameter, photoscaled *Orion* vehicle to autonomously demonstrate performance of a segmented, ablative heatshield. Currently, two TORCH flights are planned in coordination with NASA *Orion* CEV objectives. TORCH-1 will match the aerothermal shear environment under full lunar ballistic return conditions. TORCH-2 will simulate a partial aerothermal radiative heating environment under lunar ballistic return conditions and demonstrate the skip entry algorithm. The TORCH flight system also accommodates the extensive TPS instrument suite designed by NASA/Ames to evaluate the heatshield performance during re-entry.

The TORCH flight system design represents a balance between meeting customer requirements and maintaining an efficient spacecraft. Foreseeing a need to advance Entry, Descent, and Landing (EDL) technologies, the TORCH flight system is capable of accommodating different entry and cruise conditions with the intent to serve as a potential EDL test platform for other programs. By combining subsystem modularity with a common bus design, customers can minimize recurring costs in multiple flight builds. This poster gives a comprehensive overview of the TORCH flight system design, summarizing the trade studies on all subsystems, including the instrument suite accommodation and heatshield layout.

This research was carried out at the Jet Propulsion Laboratory, California Institute of Technology, under a contract with the National Aeronautics and Space Administration.

HYBRID-TPS: CURRENT DEVELOPMENTS AND PROSPECTS

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The new patented "HybridTPS" thermal protection system is under development at the NCSR "Demokritos" in Greece. The new system exploits the synergy between a rigid, high refractoriness porous ceramic and a contained reinforced ablator as a heat-sink and offers thermomechanical protection for high velocity atmospheric entries.

The HybridTPS project is funded by the European Space Agency and is expected to address some of the known problems with currently used ablative TPS, such as irregular recession of front surface leading to localised turbulence, disturbed shock wave, displaced centre of gravity and extremely brittle post-ablation char. The front surface of the HybridTPS is highly reflective and does not recess during use. The new system is produced as tiles of various shapes and sizes which are able to bond fully with each other, all but eliminating interfaces on a probe shield.

This paper will present some of the latest developments including some latest plasma test results and will discuss the future plans and prospects.